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SANTA BARBARA DIVISION

CURTISS-WRIGHT

CORPORATION - SANTA BARBARA, CALIF.

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Contract DA 44-177-TC-397

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Final Report

VZ-7AP Aerial Platform Research

SBD Report No. TR 60-37

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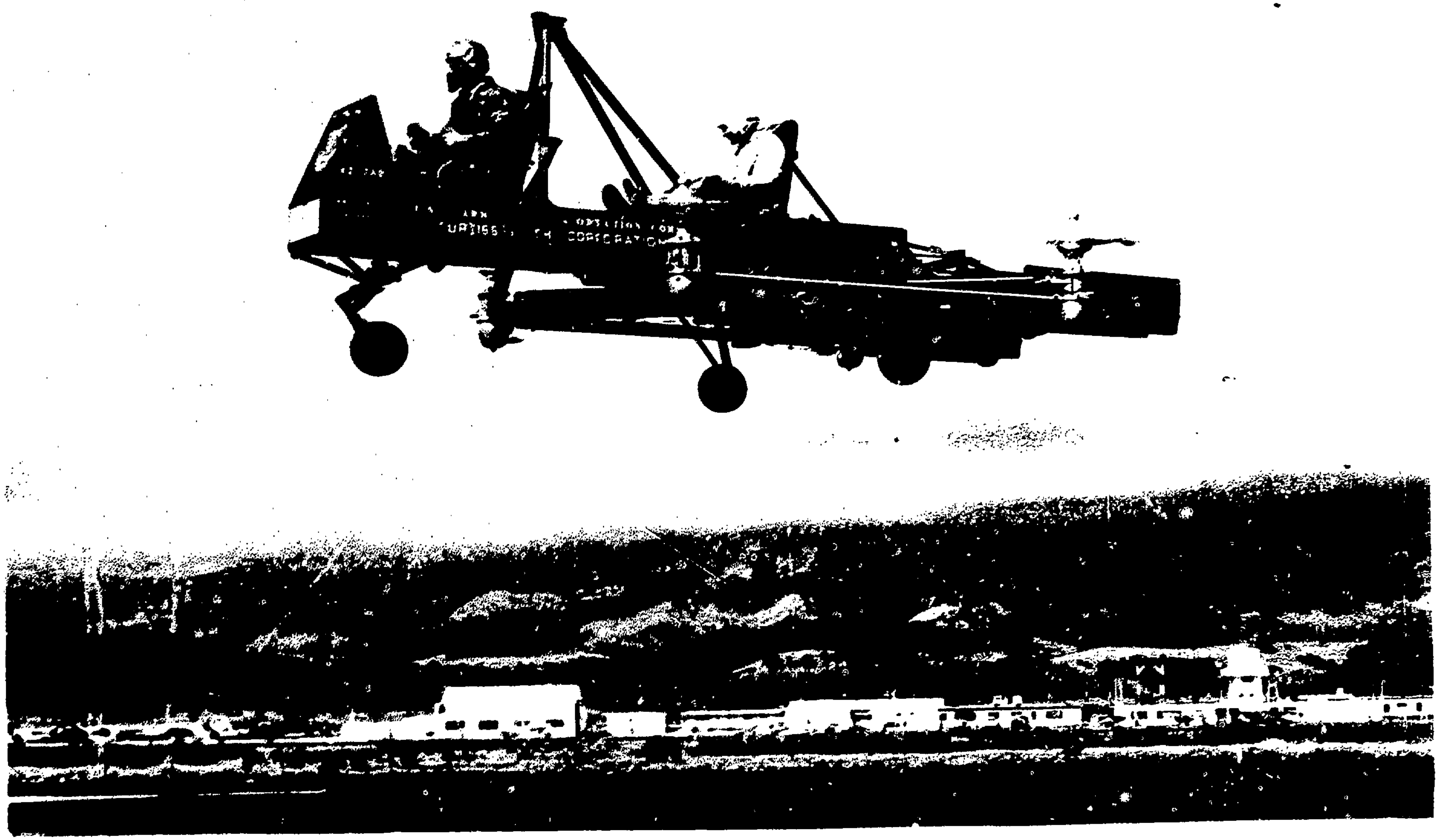

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FOREWORD

This report is submitted in compliance with the requirements of Contract DA 44-177-TC-397 between Curtiss-Wright Corporation, Santa Barbara Division, and the U. S. Army Transportation Research Command, Fort Eustis, Virginia. This contract began on 17 July 1957 and continues in force through 31 August 1960. This contract encompasses the research, analysis, design, fabrication, and testing required to determine the feasibility of a four rotor aerial jeep vehicle concept.

The activities under the contract have been divided into 3 phases: Phase I, Preliminary Design and Wind Tunnel Model Testing (Ref. 31); Phase II, Design, Fabrication and Ground Testing (Ref. 32); Phase III, Dynamic Testing and Free Flight Demonstrations (Ref. 33).

This report is the Final Engineering Report on the VZ-7AP Aerial Platform Research Vehicle.

The entire text and illustrations presented herein are UNCLASSIFIED.

PUBLICATION REVIEW

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List of Symbols

L	Lift
ρ	Air Density
A	Disc Area
V_o	Air Velocity at Disc
D	Drag
U or v	Flight Velocity
θ	Tilt of Vehicle
T =	THRUST (LIFT)
C.G.	Center of Gravity
I	Moment of Inertia
S_R	Actual Rudder Deflection
$\dot{\psi}$	Yaw Rate
K	Theoretically derived constant used in rudder effectiveness evaluation
C_y B	Fin side force coefficient as a function of yawing angle
M_u	Coefficient of pitch angular acceleration due to forward velocity
M_q	Coefficient of pitch angular acceleration due to pitch rate
.7R	Reference point on propeller for measuring blade angle
H	Height of propellers above ground
D	Diameter of equivalent single rotor

SUMMARY

This report is the Final Engineering Report on the research, analysis, design, fabrication and testing of the VZ-7AP Aerial Platform Research Vehicle, developed in response to an overall concept of an aerial jeep.

The test vehicle was originally conceived as a 4 rotor configuration with shrouds or ducts surrounding each propeller. The 4 rotor arrangement promised greater control moments than would be available with one or two rotors. The rotors were to be shrouded to take advantage of the increased thrust (lift) available.

Wind tunnel testing in Phase I established that although the shrouds would give increased lift, this advantage was more than outweighed by considerations of increased drag, increased weight, increased pitching moments and decreased vehicle controllability. The vehicle was therefore designed and fabricated using the original 4 rotor concept but with open rotors which could be surrounded with simple guard rings.

Vehicle design, fabrication and limited ground testing to establish functional and structural integrity were accomplished in Phase II.

Dynamic testing of the Aerial Platform was accomplished in Phase III in 2 steps: Tether testing and free flight testing. Tether test techniques were developed to study the dynamic behavior of the vehicle and to train the pilot without exposure to the risks of premature free flight.

Free flight testing was accomplished over a total of 102 separate flights over a period of five (5) months. Specifically, the control, stability, and performance were explored to establish the overall feasibility of the 4 rotor concept in response to the overall military aerial jeep concept.

Flight test results established that the VZ-7AP Aerial Platform possesses all of the essential characteristics envisioned for the "aerial jeep" concept.

CONCLUSIONS

The VZ-7AP has demonstrated the feasibility of the 4 rotor vehicle concept as applied to the overall military "aerial jeep" concept.

It is highly maneuverable.

Vigorous maneuvers in pitch, roll, yaw, and fixed stick stability tests from altitudes of only 10 feet demonstrated extreme maneuverability. Vehicle response was proved to be in accordance with latest NASA recommendations.

It can be flown by a relatively inexperienced pilot.

All flying was done by a pilot with very limited helicopter flight experience (although well experienced in fixed wing aircraft). His total helicopter time prior to the VZ-7AP flight program was less than 40 hours.

It is easily controlled about all 3 axes without use of stability augmenting equipment.

The inherent characteristics of the vehicle permitted successful accomplishment of every flight mission including pilot training without use of any stability augmenting equipment.

It can hover either in or out of ground effect.

Test flights were made at altitudes up to approximately 25 feet. Test data established the absence of ground effect above approximately 15 feet.

It can translate in any direction.

The VZ-7AP can fly forward, backward, sidewards or diagonally under complete control of the pilot.

It can fly at speeds comparable to ground vehicles.

The Aerial Platform was tested to 44 mph. Test environment, rather than power available, was the limiting factor in this flight test program. The nose-down tilt was only 4 degrees at this 44 mph speed.

It can carry a substantial payload.

Varying payloads of instrumentation, military loads, second crewmen and lead weights have been carried in flight tests and demonstrations. A combined useful load and payload of 647 lbs can be carried at the normal gross weight of 2400 lbs. In overload cases the vehicle has been flown at 2900 lbs gross weight.

It is both rugged and reliable.

With a very limited background of component testing (1/2 hr to 4 hrs per gear box), the Aerial Platform completed a vigorous 32 hr flight test program using the original engine, the original rotors, the original gear boxes and the original drive shafts.

RECOMMENDATIONS

The development and test activities as recorded in this report have established the feasibility of the 4 rotor Aerial Platform in satisfying the overall "aerial jeep" concept. To apply the results of this initial program to the long range objective of developing operational vehicles (as contrasted to this initial research test bed vehicle) certain areas must be probed more deeply. Curtiss-Wright, Santa Barbara Division therefore recommends the following steps be taken:

1. Full-Scale Wind Tunnel Tests

   A full-scale wind tunnel program using the original VZ-7AP Aerial Platform offers the possibilities of safely exploring regimes of performance and behavior characteristics beyond those already established in man-carrying free flight tests. The wind tunnel approach also offers improvement in instrumentation techniques making all data more exact for application to newer and more advanced vehicles.

2. Free Flight Evaluation by Government Pilots

   Such a program would offer the possibilities of verifying the free flight characteristics already established by the Curtiss-Wright, Santa Barbara Division tests, extending the flight envelope as indicated by the results of the above recommended full scale wind tunnel tests, and accumulating pilot opinion and reaction data from a number of government pilots with varying experience backgrounds.

3. Army Field Evaluation Test

   Subject the VZ-7AP Aerial Platform to a limited field evaluation test wherein the feasibility of the jeep concept can be established in the Army field environment. Such factors as tactical employment, dust raising characteristics, noise, logistic support and field operation in general could be explored and evaluated towards the establishment of military characteristics for future vehicles.

Curtiss-Wright, Santa Barbara Division further recommends that the accumulated knowledge from the present program herein reported, and from the steps recommended above, be applied to the development of operational configurations of 4 rotor military vehicles. The 4 rotor concept is not limited to the aerial jeep concept. It can be applied to a variety of vehicle sizes and specific military tasks. Such vehicles must be developed around practical requirements generated within the potential using agencies and expressed in desired military characteristics.

1. INTRODUCTION

Expanding concepts of required mobility within the U. S. Army in the period 1956-1957 led to the establishment of what has been termed the "aerial jeep" concept. The Army recognized the basic utility of the popular ground jeep and wished to develop an airborne vehicle which could accomplish many of the tasks of the ground jeep and provide 3 dimensional mobility to accomplish tasks that earth-bound vehicles could not accomplish. The desired vehicle was to fly low and slow, following ground contours in and out of gullies, hiding under trees, and taking full advantage of the terrain. Additionally, the vehicle should have the ability to fly out of ground effect to permit reaching otherwise inaccessible points such as mountain peaks.

As a result of military interest in this new vehicle concept, technical proposals were solicited from the aviation industry by the U. S. Army Transportation Research Command (TRECOM), Ft. Eustis, Virginia. The Army recognized the existence of many problem areas and allowed the bidders wide latitude in choice of vehicle configurations.

As a result of these proposals, contracts were consumated in July 1957 for research test bed vehicles from which basic principles could be developed, and from which basic data and experience from flight tests could be gathered.

The 4 rotor aerial platform configuration was conceived by Curtiss-Wright Corporation, Santa Barbara Division in the precontract studies. It was studied, developed and proven during the course of research contract DA 44-177-TC-397.

For purposes of evaluation and control, the contract was divided into 3 distinct phases, each with its own specific objectives.

Phase I	Preliminary Design and Wind Tunnel Model Testing. This work was accomplished between July 1957 and December 1957.
Phase II	Design, Fabrication and Ground Testing. This work was accomplished between December 1957 and July 1959.
Phase III	Dynamic Testing and Free Flight Demonstrations. This work was accomplished between August 1959 and January 1960.

Individual month-by-month progress in study, design, fabrication and testing was recorded in Curtiss-Wright Corporation, Santa Barbara Division monthly progress reports, references 1 thru 30.

Final Reports for each of the 3 contract phases were submitted to TRECOM with supporting analyses, test results, etc., references 31 thru 33.

This report is the Final Engineering Report under the contract, and highlights the activities of the entire program, with particular emphasis on the results obtained. This Final Report is arranged in chronological order with minor variations to permit grouping of related subject material.

2. DESIGN DATA

PRELIMINARY DESIGN CONSIDERATIONS

The basic 4 rotor configuration for the Aerial Platform was chosen after careful consideration and comparison with 2 and 3 rotor configurations. The 4 rotor configuration with controllable pitch propellers offered distinct advantages in regards roll and pitch control moments, cancellation of gyroscopic couples and large disc area within specified width limitations.

After the contract was awarded, Curtiss-Wright engineers made basic layouts of the configuration to ascertain the major problem areas. The gear system was one that would require specialized engineering to obtain a light weight transmission, capable of handling in excess of 425 horsepower and which would have such inherent reliability to permit flight testing without extensive bench testing.

The propellers and shrouds were another problem area. A search of the available literature on shrouded propellers with relatively short shrouds did not produce applicable data for the case of the shrouded propeller moving sideways thru the air.

Certain aspects of the vehicle structure were considered to be unusual and would require special attention. These included the problems of (1) vibration and fatigue loads setup by the propellers rotating in close proximity to the structure and of (2) structural rigidity for necessary gear alignment.

Stability and control were recognized as problem areas, and would require special study.

Gear System Considerations

The requirement to drive propellers from a single engine could have been solved by several different gear box and shaft arrangements. In their original proposal prior to award of contract, Curtiss-Wright engineers had established the desirability of using a cruciform layout of shafts, wherein each propeller gear box was fed from a central gear box by means of intermediate drive shafts. This arrangement promised a minimum number of different gears and gear boxes and a maximum interchangeability of basic elements.

A parametric study of the total gear system was made to determine propeller rotational speeds and intermediate shaft speeds for optimum weight, based on a propeller tip speed of 750 feet per second and an engin e output shaft speed of 6000 RPM. The results of this study are shown in Figures 1 and 2.

In regards propeller rotational speeds (Figure 1), it can be seen that the total gear system weight goes down as the propeller rotational speed goes up, however the major economy in weight is realized by operating in the vicinity of 2400 RPM, which was chosen as the index point for the study.

In regards the rotational speed of the intermediate drive shafts (Figure 2) it can be seen that for a propeller rotational speed of 2400 RPM, the optimum weight is achieved at a drive shaft speed of approximately 4000 RPM.

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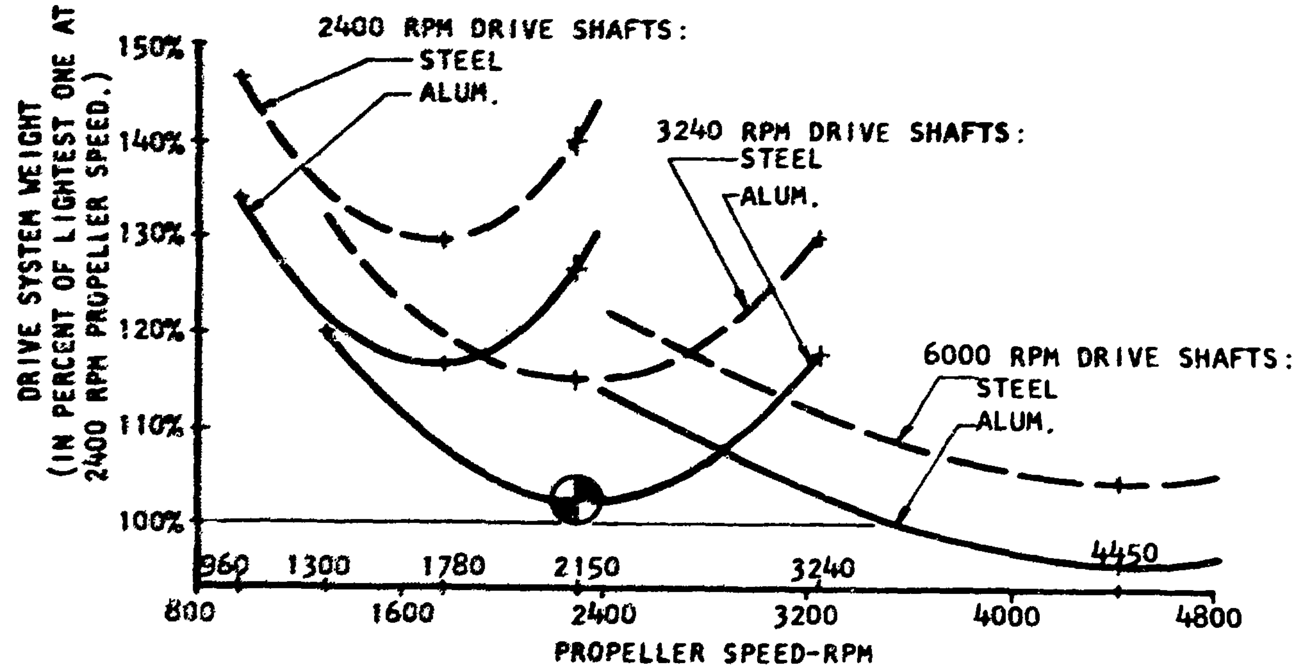
DRIVE SYSTEM* WEIGHT VS. PROPELLER SPEED



* DRIVE SYSTEM CONSISTS OF:
1 MAIN GEAR BOX
4 PROPELLER GEAR BOXES
4 DRIVESHAFTS (ALUM. ALLOY OR STEEL AS NOTED)
AERIAL PLATFORM PROPELLER & SHAFT RPM

Figure 1 DRIVE SYSTEM WEIGHT VS PROPELLER SPEED

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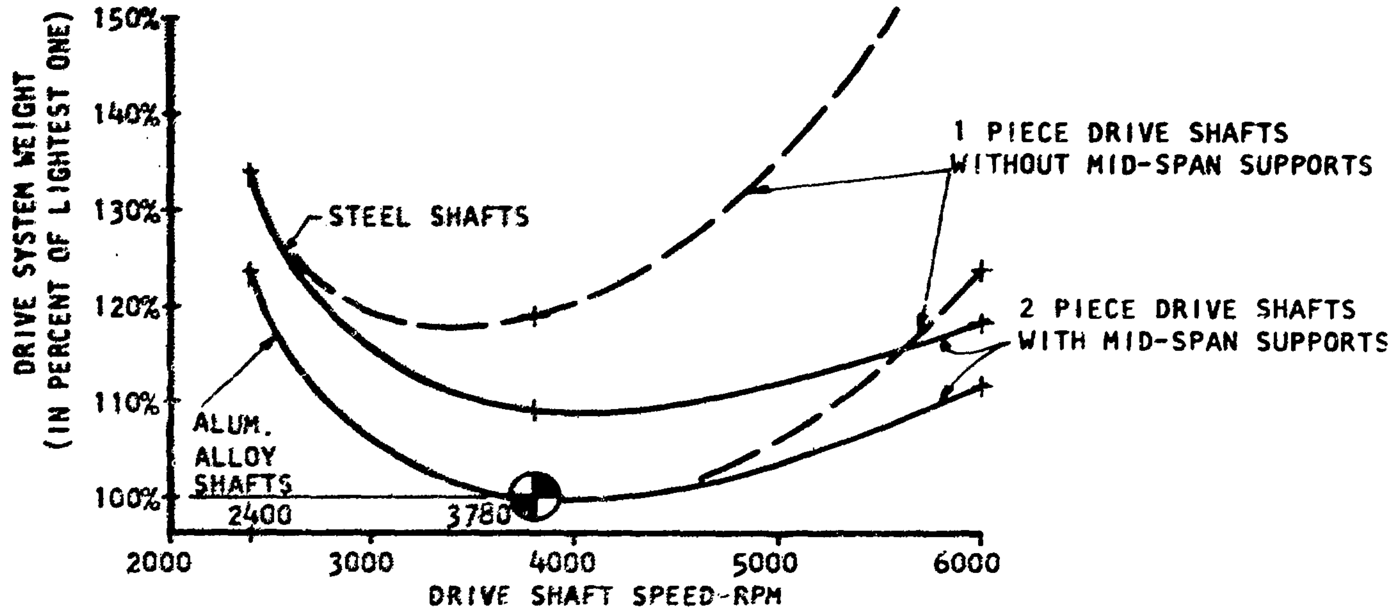
DRIVE SYSTEM* WEIGHT VS. DRIVE SHAFT SPEED
(PROPELLER SPEED = 2400 RPM)



*DRIVE SYSTEM CONSISTS OF:
1 MAIN GEAR BOX
4 PROPELLER GEAR BOXES
4 DRIVE SHAFTS (ALUM. ALLOY OR STEEL AS NOTED)
AERIAL PLATFORM PROPELLER & SHAFT RPM

Figure 2 DRIVE SYSTEM WEIGHT vs DRIVE SHAFT SPEED

Propeller and Shroud Considerations

A series of wind tunnel tests was carried out to determine configurations which might be suitable. That is, a configuration was sought such that the vehicle would have sufficient power available so that its lift may be made equal to weight and thrust equal to drag at all the desired operating conditions. In addition, sufficient trim moments should be available to balance out the moments which might arise due to eccentric loading of the vehicle or operating in forward flight, climb, descent, or hovering.

The dependence of lift, drag, and pitching moment on the shroud geometric parameters (a) entrance lip radius, (b) shroud length, (c) propeller position along the axis of the shroud, and (d) divergence of the shroud downstream of the propeller, were examined. See Table 1.

Detailed dependence of lift, drag, pitching moment, and power on the parameters (a) flight velocity, (b) angle of attack, and (c) propeller blade angle were examined for three representative configurations; a propeller alone and a propeller in two typical shrouds.

The nature of the forces on a typical duct were studied by examination of a series of pressure distributions. Certain assumptions made in designing the propeller were examined for validity in a series of flow survey pictures.

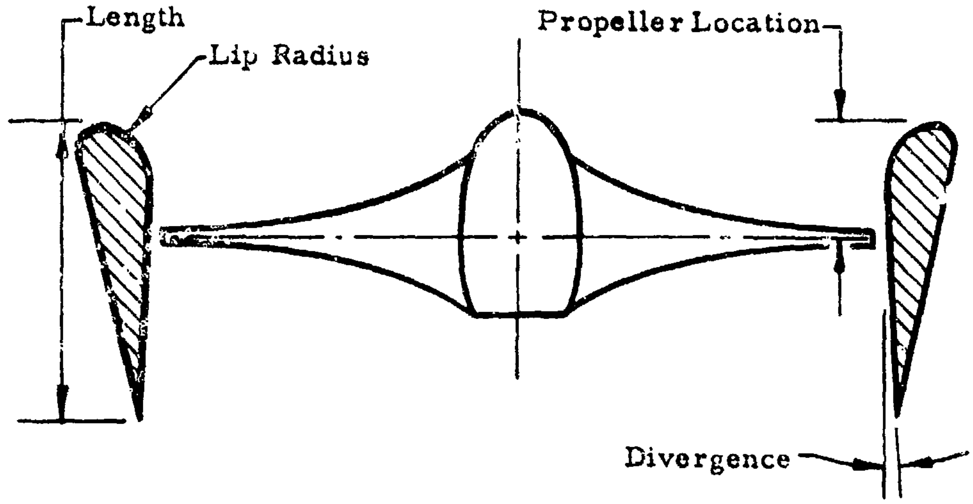
Devices which might be desirable for improving the basic propeller shroud combination were examined.

For the preliminary studies it was chosen to ignore temporarily the interference between rotors of the vehicle and examine the characteristics of a single rotor. Later tests used two rotors to determine mutual interference effects.

A one-tenth scale model of a propeller and various ducts and special devices were built and tested in the subsonic wind tunnel at the University of California at Los Angeles. The model size was chosen primarily by what would be acceptable in the tunnel. Table 2 compares full scale and model properties. The model is shown in Figure 3.

TABLE 1

Shroud Parameters
Investigated in Model Tests



Shroud Identification	Shroud Length*	Inner Lip Radius*	Divergence Angle	Propeller Locations*
A	20%	6.7%	0°	16.6%
B	30%	6.7%	0°	16.6%, 28.5%
C	40%	6.7%	0°	16.6%, 28.5%, 36.8%
D	40%	6.7%	4°	16.6%
E	30%	3.3%	0°	16.6%, 28.5%
F	30%	10.0%	0°	16.6%, 28.5%
G	30%	10.0%	4°	16.6%
H	30%	10.0%	0°	16.6%
I	30%	15.0%	0°	15.0%

*All dimensions are given as percentages of the propeller diameter which was 6.00 inches.

TABLE 2

COMPARISON OF FULL SCALE AND MODEL PROPERTIES

Configuration	Full Scale	Model
Number of Rotors	4	1 or 2
Rotor Diameter, ft.	6	0.6
Tip Speed, ft./sec.	754	754
Rotational Speed, revolutions per minute	2,400	24,000
Lift or Drag Force (typical) lbs.	2,000	5 per rotor
Pitching Moment (typical) inch pounds	10,000 per rotor	10 per rotor
Power (typical), hoursepower	400	1 per rotor
Flight Speed, ft./sec.	85	85
Tip Mach Number (hovering)	0.67	0.67
Tip Reynolds Number (hovering)	2.41×10^6	0.241×10^6

Note that the velocity, Mach number, and pressure will be equal at corresponding points of the model and full scale vehicle. Pressure gradients on the model will be ten times those on the full scale vehicle and Reynolds numbers will be one tenth of those on the full scale vehicle.

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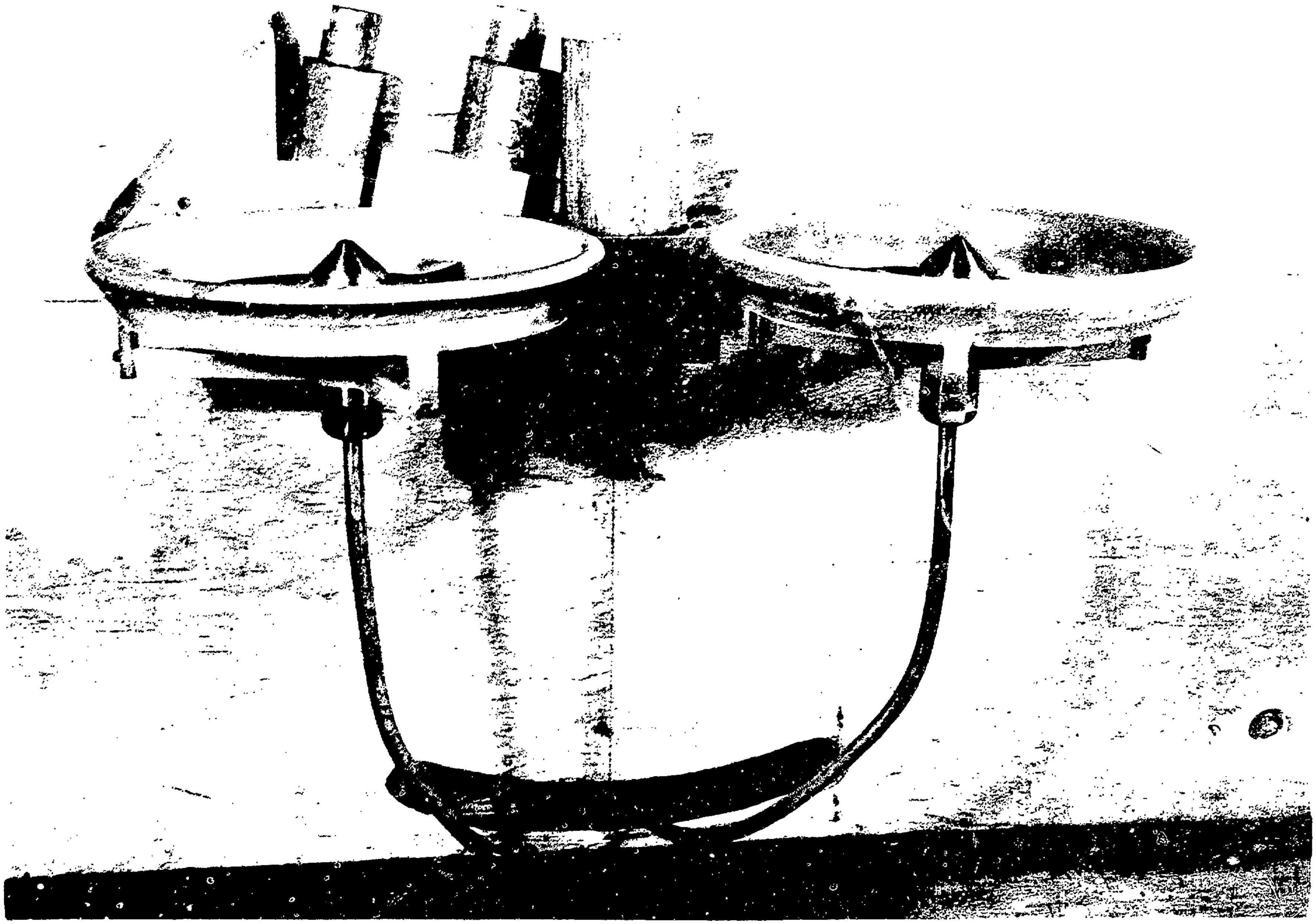


Wind Tunnel Test Results

The results of selected tests are given below to illustrate certain trends which were observed.

Lift: (See Figure 4)

The addition of a suitable shroud to an existing propeller improved the lift per horsepower in hovering flight. The principal effect of the addition of the shroud was to increase the slope of the lift per horsepower curves. All of the shrouds tested had approximately the same slope, the effects of varying the shroud parameters being to shift the curves parallel to each other. Shrouds with larger inner lip radii and/or longer length produced more lift per horsepower. The lift decreased with increased depth of propeller inside the shroud.

The increase in lift per unit horsepower for all combinations of shroud parameters tested varied from 11% to 26% over that produced by the bare propeller.

An increase of 4% was found using only a simple guard ring rather than a shroud.

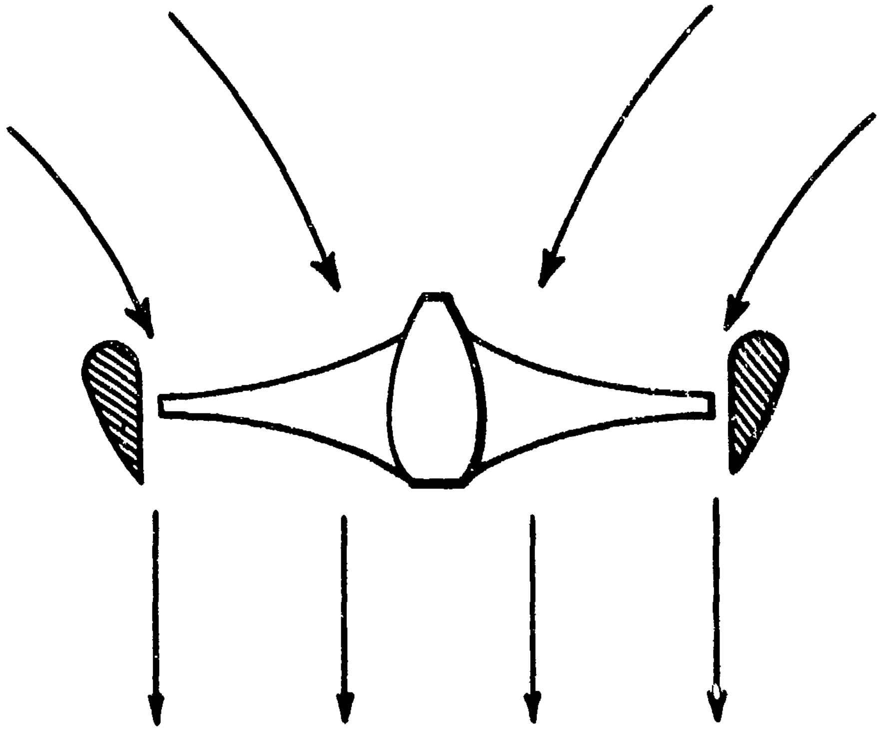
Drag: (See Figure 5)

The addition of a shroud to the existing propeller sharply increased the drag per unit lift in translational flight. This is a fundamental problem in the use of shrouded propellers in forward flight; the shroud tends to cause the airflow through the propeller to be parallel with the propeller axis, thus effectively removing the horizontal momentum of the air traversing through the duct.

The drag per unit lift obtained under conditions of zero angle of attack and flight velocity of 97 feet per second was found to be reasonably independent of entrance lip radius but to depend quite strongly on shroud length.

Under these test conditions the drag per unit lift for the shroud and propeller was found to vary from 217% to 238% of that for the bare propeller alone.

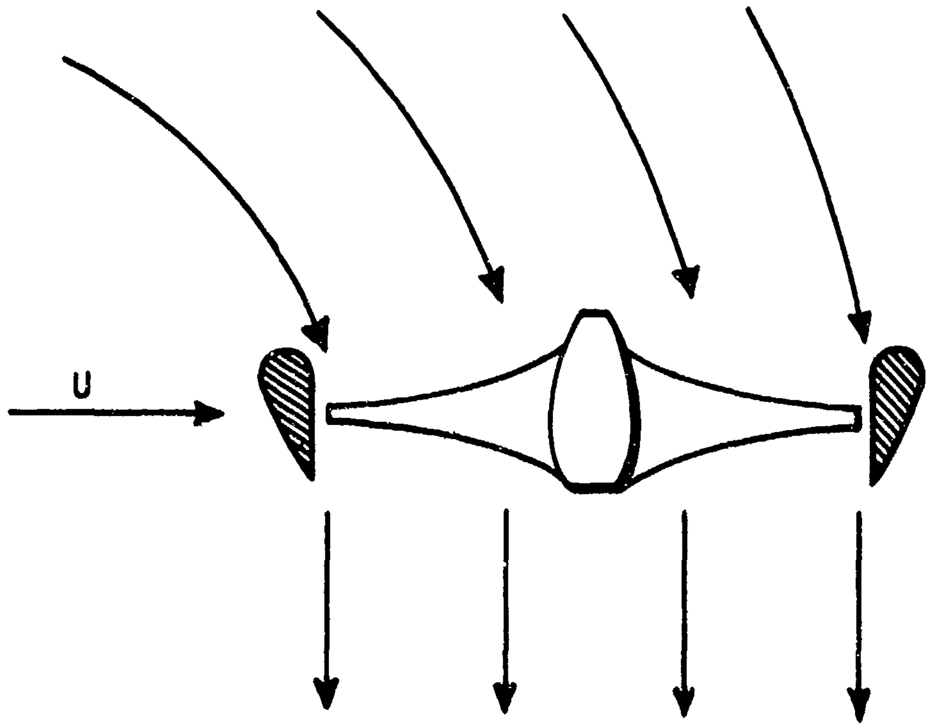
On this same comparative basis, the drag per unit lift using shrouds of different length varied from 193% to 256% with increases up to 279% where the propeller was submerged more deeply in the shroud.



$$L = (\rho A V_o)\, V_o$$

L ~ LIFT
ρ ~ AIR DENSITY
A ~ DISC AREA
V_o ~ AIR VELOCITY AT DISC

Figure 4 APPROXIMATE LIFT, HOVERING



$$D = (\rho A V_o)\, U$$
$$= \frac{U}{V_o} L$$

D ~ DRAG

U ~ FLIGHT VELOCITY

Figure 5 APPROXIMATE DRAG IN FORWARD FLIGHT

Pitching Moment

Shrouded propellers develop large pitching moments in translational flight. At the test velocity of 97 feet per second the pitching moment was found to be reasonably independent of the shroud length but greatly dependent on inner lip radius of the shroud and propeller depth within the shroud. This indicates that a considerable part of the pitching moment was caused by the drag load on the duct lip since both the lift and drag were practically independent of the propeller position within the shroud.

The pitching moment produced by changing the propeller depth within the shroud was found to be frpm 252% to 326% of the moment produced by the bare propeller. These values correspond to propeller depths of 16.7% and 28.5% of the propeller diameter in a shroud whose length was 30% of the propeller diameter.

The pitching moment for shrouds with different lip radii varied from 148% to 428% of the moment produced by the bare propeller. These broad values were produced by ducts having lip radii of 3.3% and 14.2% respectively of the propeller diameter.

A representative pressure distribution on the shroud in translational flight is shown in Figure 6. The net force on the propeller shroud combination can be seen to be approximately 25% of the propeller diameter ahead of the propeller axis, thereby producing a pitching moment of considerable magnitude.

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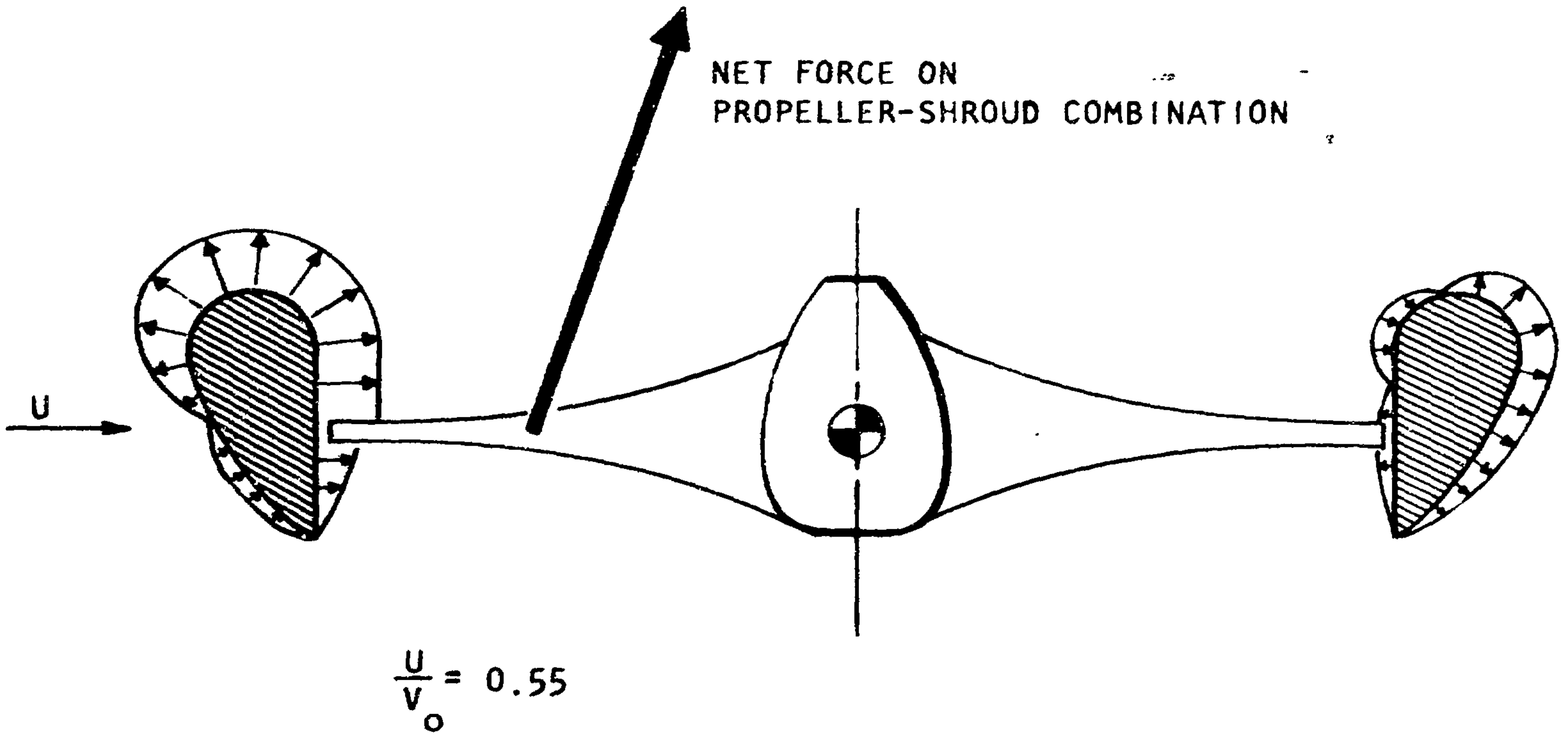


Figure 5 PRESSURE ON SHROUD IN FORWARD FLIGHT

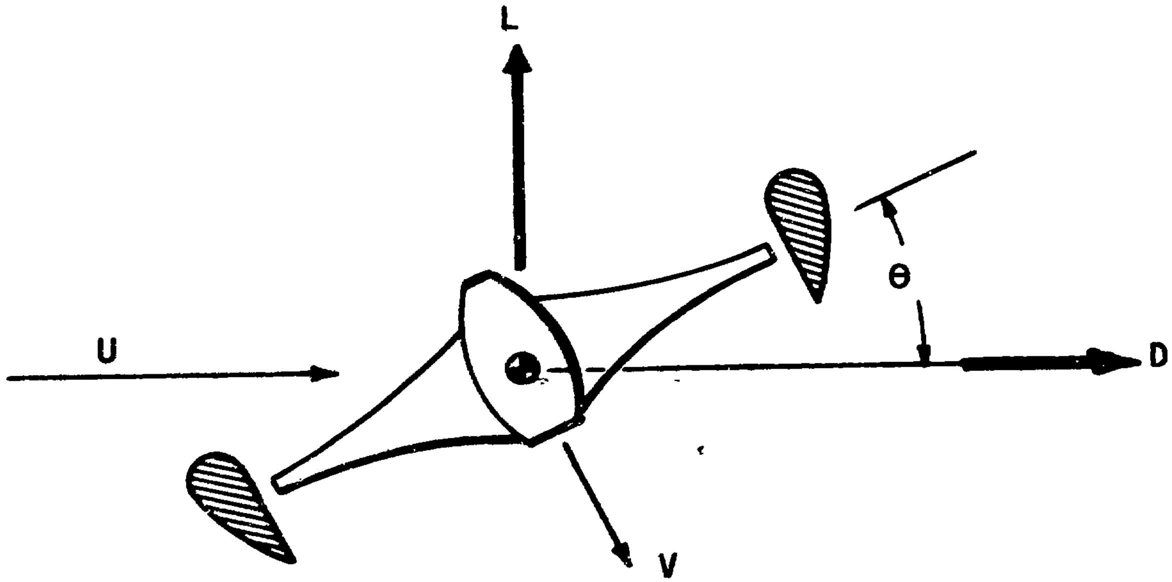
Thrust:

In order to fly a vehicle using shrouded propellers the net drag must be zero. One way of accomplishing this would be to tilt the shroud leading edge down (Figure 7).

Since lift must be kept equal to weight and drag made zero for steady horizontal flight, the expressions may be solved to obtain the tilt angle required for a given flight velocity. As shown in Figure 8 the required tilt angle for the design maximum speed (50 knots) would be very large (40°).

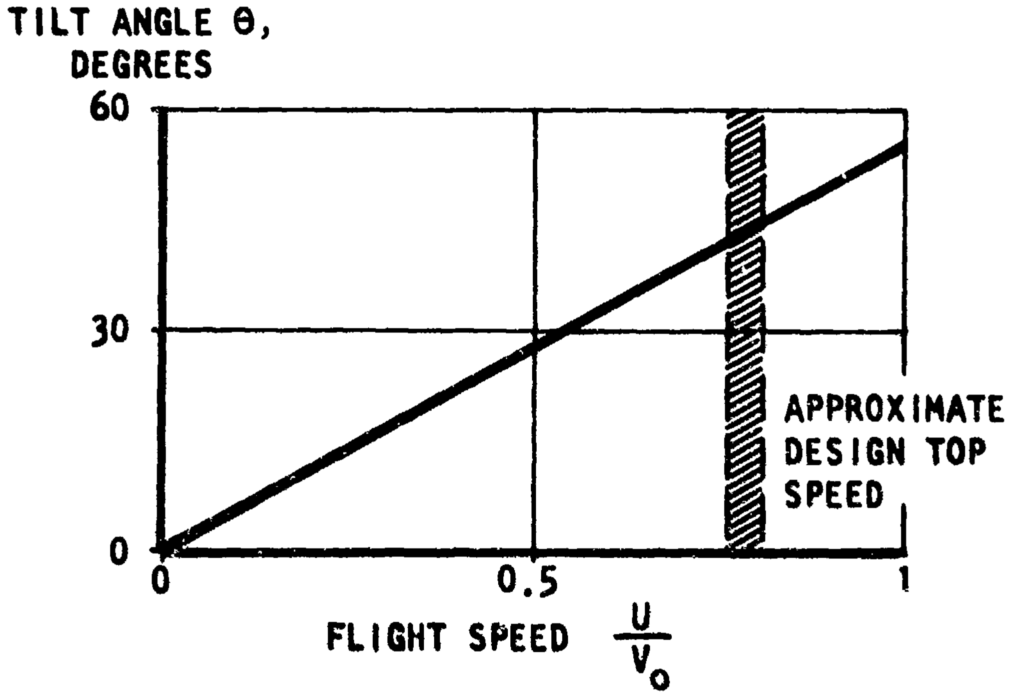
A second method of producing forward thrust with a shrouded propeller would be to use a cascade of turning vanes under the propeller (Figure 9). In hovering the blades could direct the air downwards; in forward flight the air could be directed rearward.

The wind tunnel tests showed that the thrust can be made equal to the drag in forward flight at velocity of 40 feet per second at zero angle of attack. The forward velocity attainable in flight at zero angle of attack was shown to depend on the angle that the cascade blades make with the propeller axis. The cascade results in a loss of lift per horsepower because of aerodynamic drag on the blades, especially at high deflection angles; and results in a loss of net vehicle lift because of the weight of the vanes.



$$L = (\rho AV)\ V \cos \theta$$
$$D = (\rho AV)\ (U - V \sin \theta)$$

Figure 7 APPROXIMATE LIFT & DRAG IN INCLINED FORWARD FLIGHT



$$V_o = \sqrt{\frac{\text{HOVERING LIFT}}{\rho A}} \qquad U = \text{FLIGHT SPEED}$$

Figure [illegible] NOSE DOWN TILT vs FORWARD SPEED

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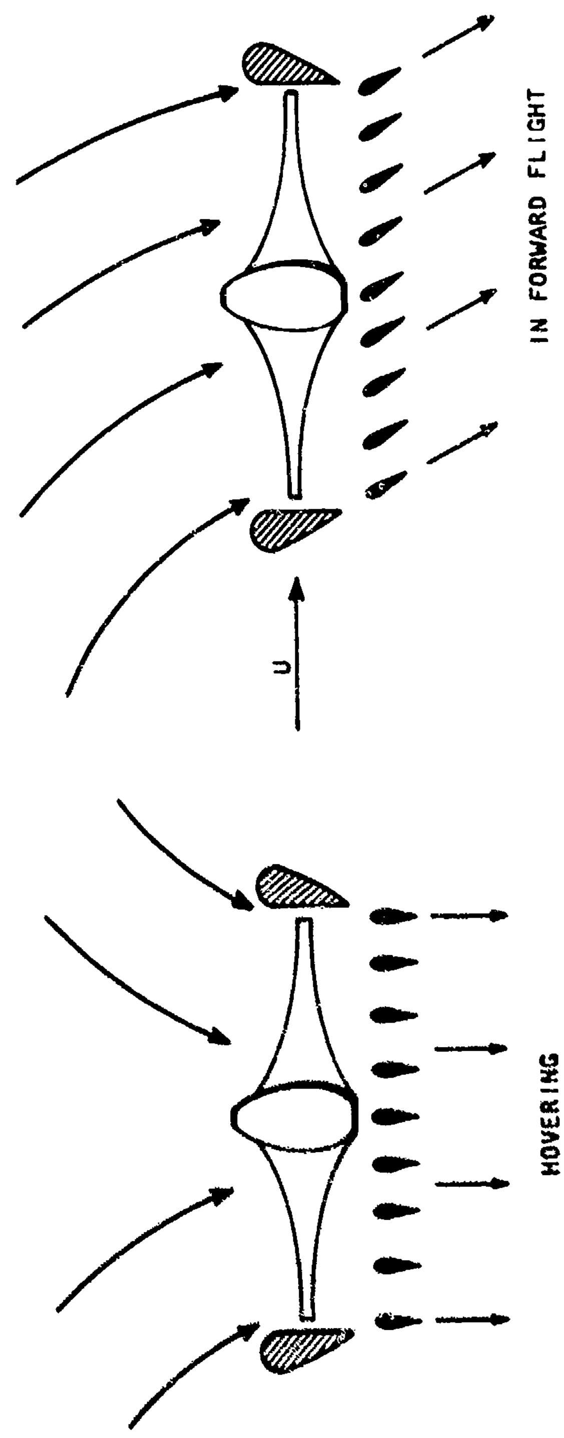

Figure [illegible] SHROUDED PROPELLER WITH TURNING VANES

Wind Tunnel Test Evaluation:

Upon examination of the magnitudes of the various phenomena mentioned above (based on theoretical and experimental evidence), it was decided that the purposes of the program could best be accomplished by building a vehicle having only the most rudimentary shrouds (that is, guard rings). The following factors had large influence on the decision:

(1) A guard ring around the propeller (a shroud of very small length) has still a beneficial effect on the hovering thrust per horsepower (about a 4% improvement).

(2) A thin shroud would permit the propeller diameter to be increased, resulting in an increase of hovering thrust per horsepower because of the lower disc loading.

(3) The drag in translational flight would be low because the propeller has less effect on the momentum of the air passing through the disc than the duct.

(4) The pitching moments on the unshrouded rotor in forward flight were considerably smaller than for a heavily shrouded propeller.

Propeller Pitching Moments

The size, rotational speed, and power requirements of each propeller on the Aerial Platform were quite similar to existing controllable pitch propellers for normal airplanes. The existence of proven propeller hardware and techniques would reduce the number of unknowns in testing the end article. However, certain differences existed in the manner in which the normal airplane propeller is loaded and the manner in which it would be loaded on the Aerial Platform.

A propeller similar to an airplane propeller with fixed blade pitch, experiences rolling and pitching moments in edgewise translational flight. These forces are reduced to manageable magnitudes in conventional helicopters by hinged blades and cyclic pitch control. Upon examination of the magnitude of the loads it was determined that the propeller blades for the Curtiss-Wright Aerial Platform should have flapping hinges. The bending stresses in the rotor, as well as the pitching and rolling moments of each rotor, are in this way reduced to a minimum. The remaining moments result from the difference in load between the advancing and retreating blades, and the distance the blade hinges are offset from the shaft. The flapping hinges also eliminated large oscillatory loads on the support arms and transmission system in translational flight.

Calculation of these moments takes into account the aerodynamic forces on the blade and the inertial forces due to flapping action. Figure 10 shows typical results for a rotor in forward flight at an advance ratio (flight speed to propeller tip speed) of 0.12 and 800 pounds of thrust. The figure shows the variation in pitching moment, rolling moment, and the resultant moment.

By rotation of the right and left hand propellers in opposite directions, and phasing of the rotors, the rolling moments cancel each other in normal forward flight. The pitching moments on all four rotors are additive, however, and require the transfer of more and more thrust to the aft propellers as the platform goes from hovering to flight in the forward direction. Figure 11 shows typical thrust and moment distributions for hovering flight (μ = o) and forward flight (μ = 0.12) for a forward center of gravity position. Figure 12 shows the manner in which the pitching moments vary with thrust, other conditions remaining constant.

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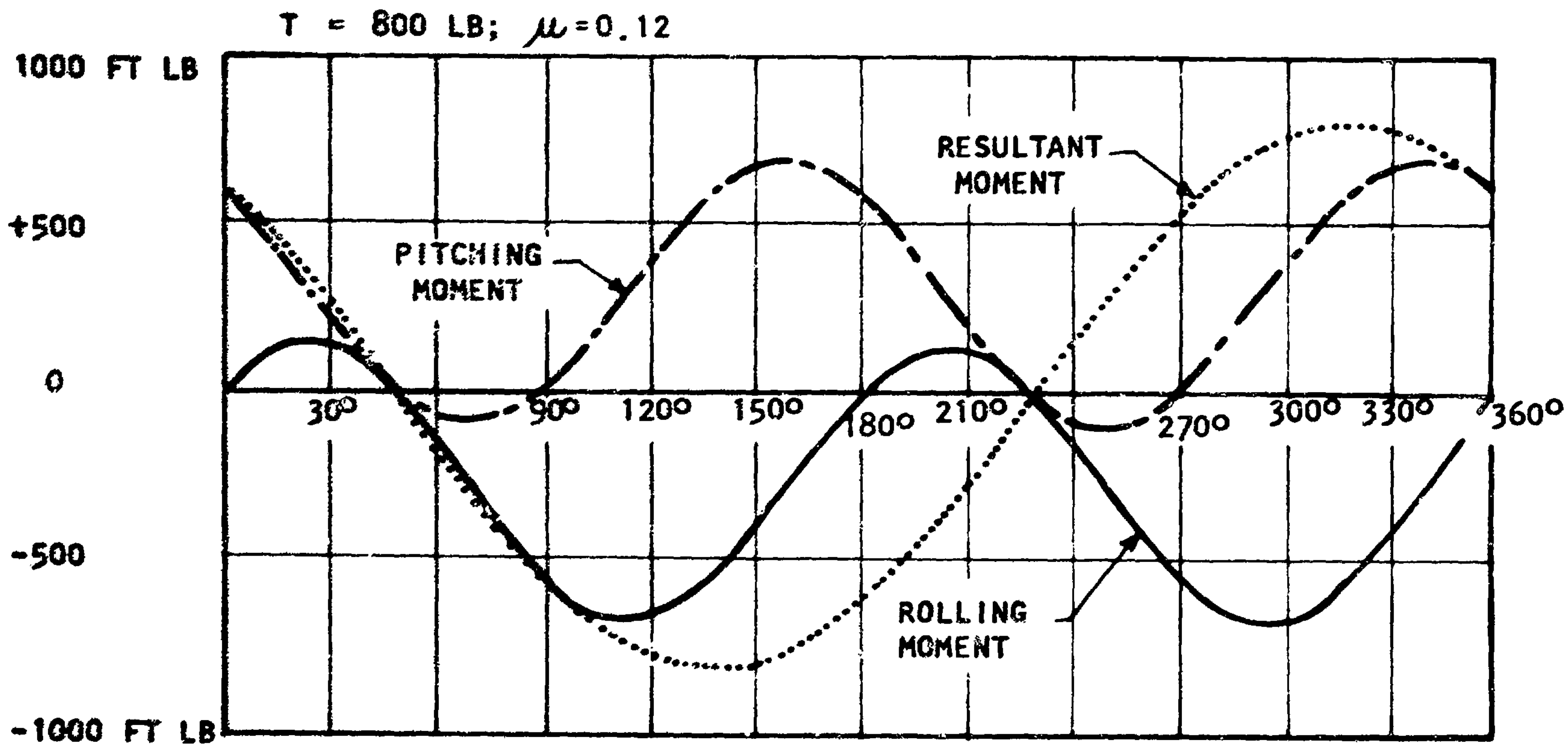

Figure 16 MOMENTS AT PROPELLER HUB

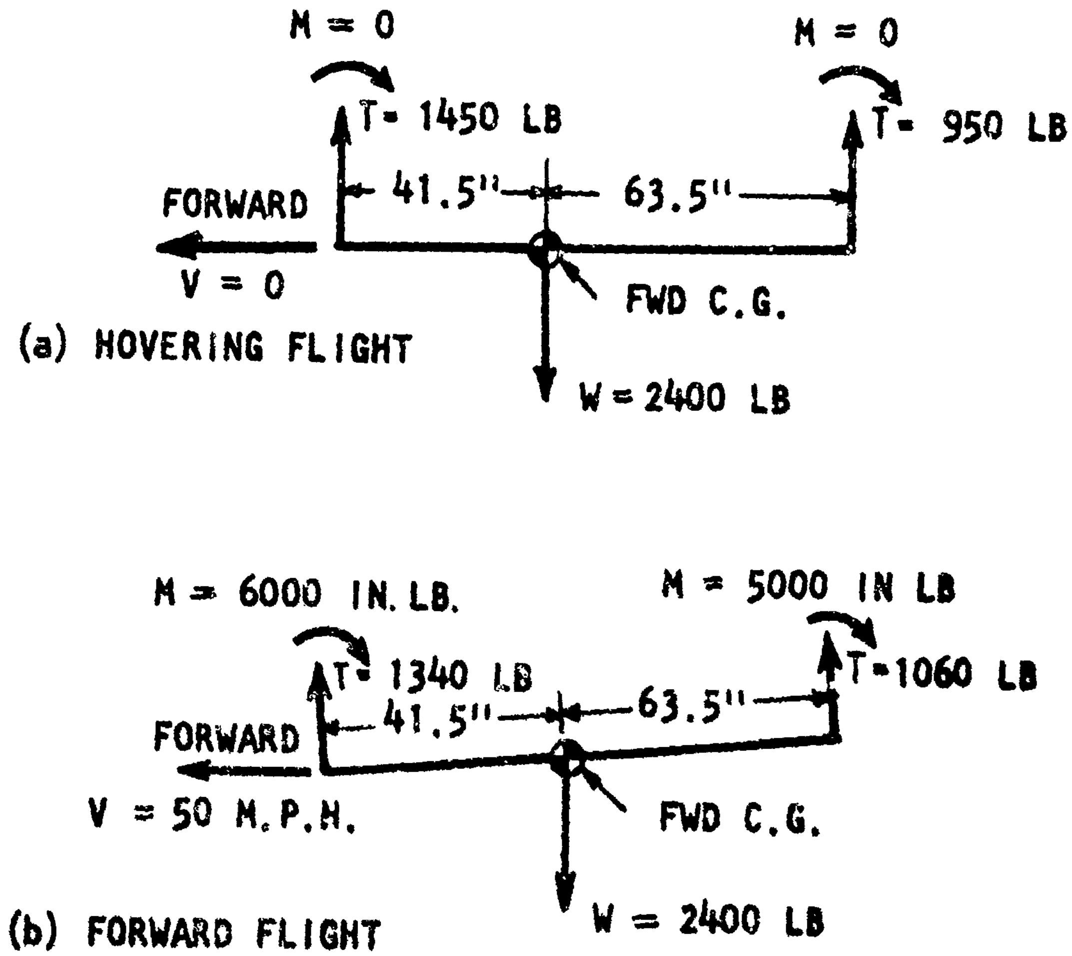

Figure 11 TYPICAL THRUST & MOMENT DISTRIBUTIONS

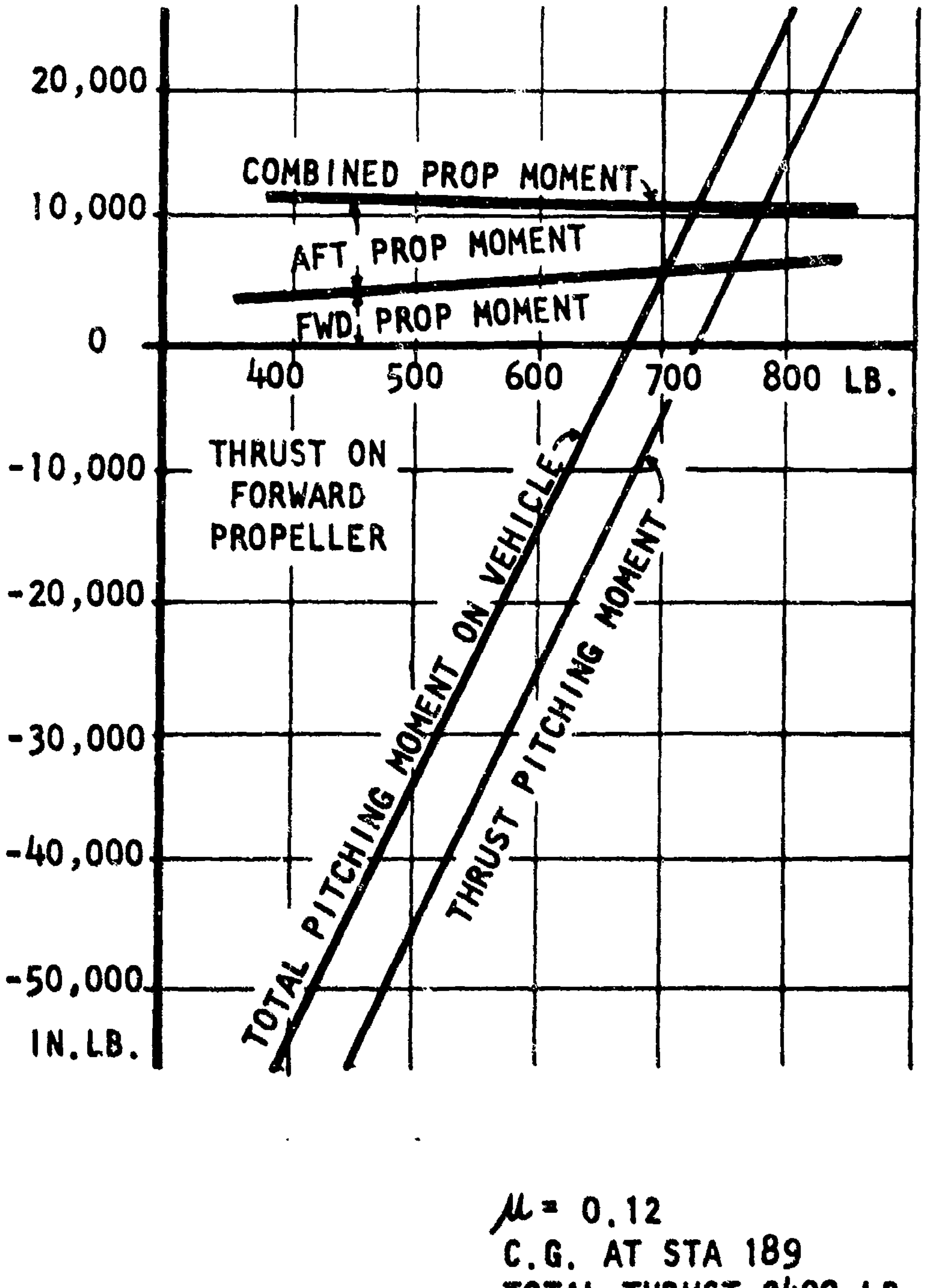

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Figure 12 AVERAGE PITCHING MOMENTS

Airframe Structural Considerations

The airframe structural configuration which was chosen in preliminary design is a simple rectangular cross-section structure utilizing extruded aluminum alloy longerons and flat aluminum alloy shear panels.

The anticipated problems of vibration and fatigue associated with the propellers operating so close to the basic structure led to the selection of bonded aluminum alloy honeycomb sandwich panels, for the sandwich panels offered unusually high rigidity and damping. An additional benefit was that this type of structure could be assembled with a minimum of jigs and tools.

The success of the gear drive system would to a large degree depend upon accurate alignment and rigid mounting of the gear system elements with respect to each other. Therefore, it was determined that the gear system would be built as a rigid frame within itself, further stiffened by the basic structure to which it is attached. (Figure 13)

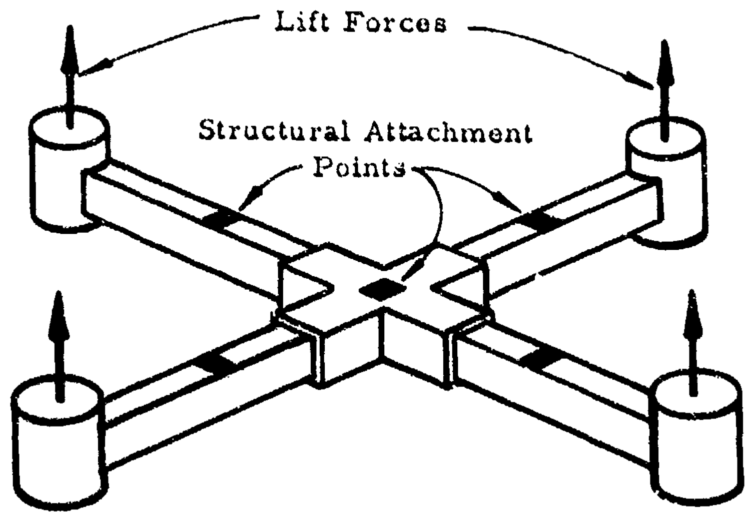


Figure 13 RIGID FRAME CONCEPT FOR THE PROPELLER SUPPORT SYSTEM

Stability and Control

An analytical study revealed that the Aerial Platform would be difficult for the pilot to control because of a divergence in pitch and roll, having a period of about 5 seconds. To overcome this difficulty, stability augmentation by artifical means was considered necessary.

Stability and control studies were continued using an analog computer representation of the vehicle motion in pitch (Figure 14). The analog studies confirmed the earlier findings that the pilot would find it difficult to fly the vehicle without assistance from a stability augmentation system. For example, on the analog set-up a typical pilot was instructed to tilt the vehicle to a nosedown attitude of six degrees from hovering position, maintain the tilted position and then return to the hovering position. The pilot was able to stabilize the vehicle, though he found the task to be mentally very fatiguing. The fact the pilot had to read and intrepret instruments rather than experience any position feel feedback of course complicated the problem of "flying" the vehicle on the analog computer.

One important characteristic was established in these studies: The Aerial Platform must be flown by controlling its tilt attitude. Every attempt to fly it by velocity failed on the analog computer. Analytical examination of the problem established that a system of rate damping would be adequate for stability augmentation. This was readily confirmed by mechanization of the control equation on the analog computer. The earlier flight problems which had been so difficult without stability augmentation were repeated with excellent control of both attitude and velocity with the rate damping stability augmentation system working.

The specific details of the stability and control study are presented in reference 31.

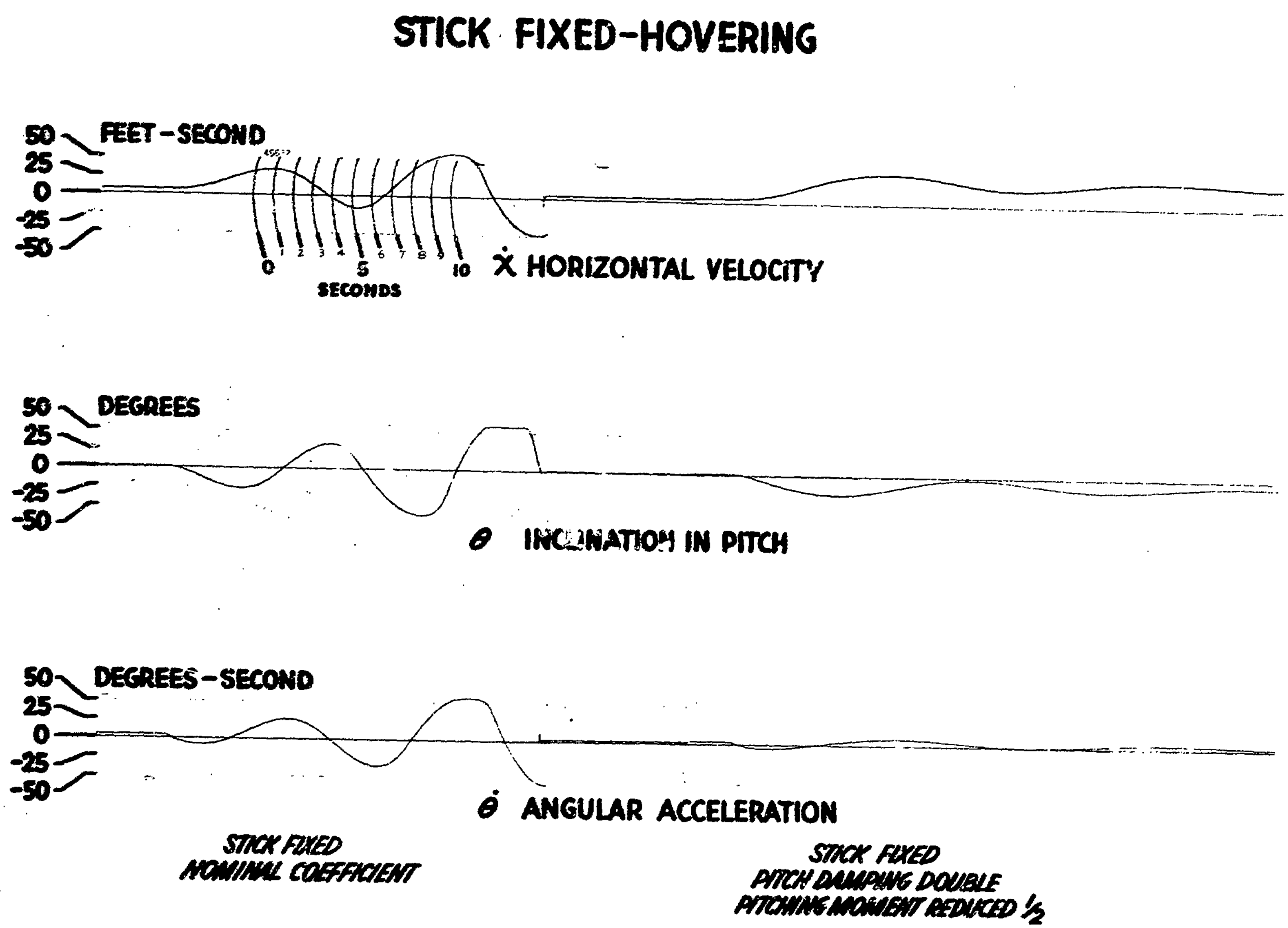


Figure 14 ANALOG STUDY, FIXED STICK RESPONSE

3. DESCRIPTION OF TEST ARTICLE

BASIC CONFIGURATION

The basic configuration of the airframe as shown in Figures 15 and 16 had been established as a result of the Preliminary Design activities accomplished in Phase I, reference 31. The configuration featured:

a. 4 un-shrouded propellers, with offset flapping hinges

b. Low overall configuration

c. Flat cargo deck at convenient loading height

d. Box-like structure

e. Rigid frame propeller support system independent of basic structure

f. Tricycle landing gear

g. Dimensions to suit loading width of USAF C-130 transport (See Figure 17)

A summary of vehicle basic data is presented in Table 3.

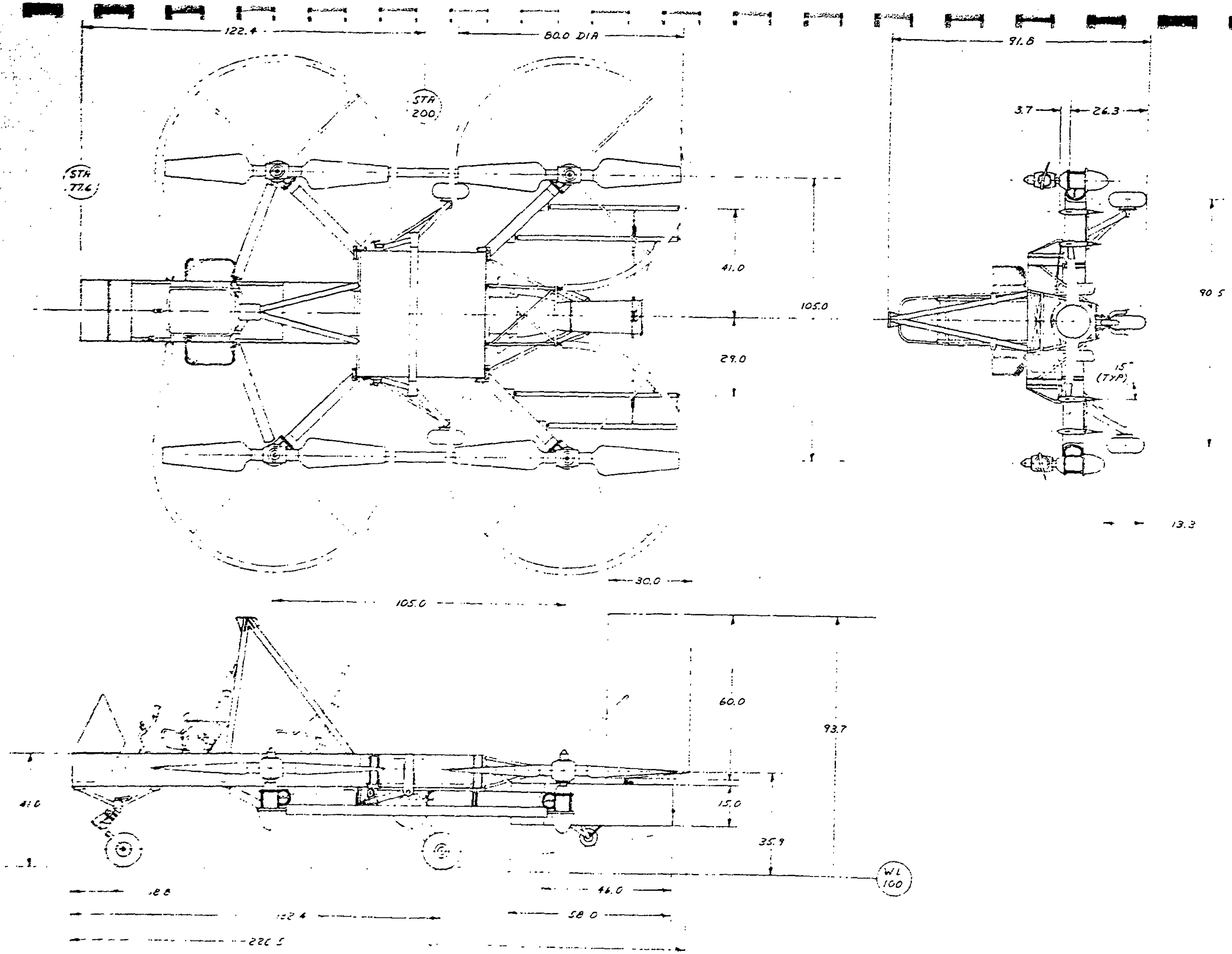
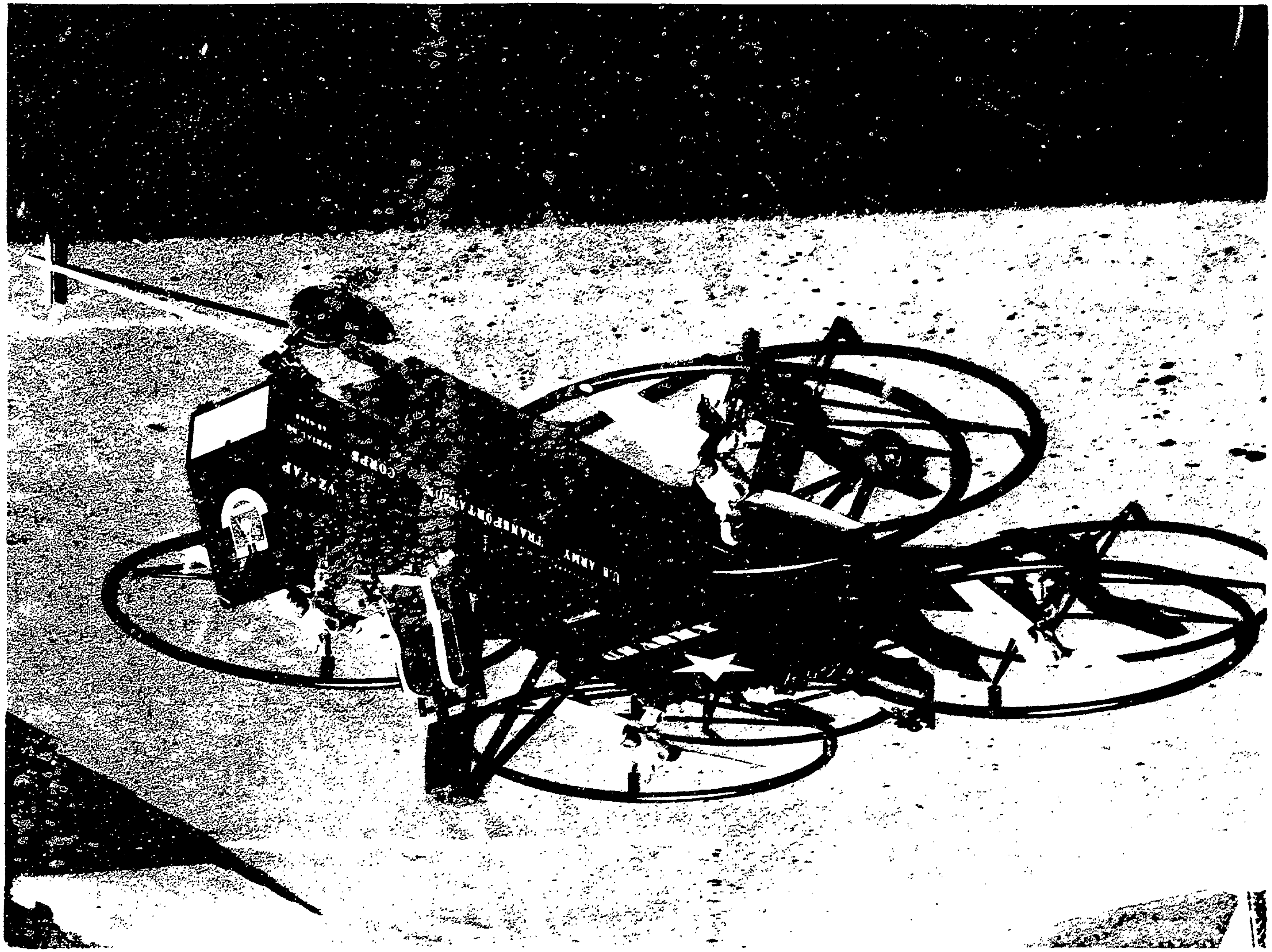


Figure 15 GENERAL ARRANGEMENT DRAWING VZ-7AP AERIAL PLATFORM

TABLE 3

GENERAL DATA

Gross Weight (Design)	2400 lbs
Propeller Diameter	6.5 feet
Propeller Disc Area (Total)	133. sq. st.
Disc Loading	18. lbs/sq. ft
Engine Power	400. h.p.
Power Loading	6.0 lbs/h.p.
Engine Speed - Turbine Wheel	34,800 RPM
Engine Speed - Output Shaft	6,000 RPM
Gear Reduction, Main Gear Box	17/27
Gear Reduction, Propeller Gear Box	19/33
Propeller Rotational Speed	2175 RPM
Propeller Tip Speed	740 ft/sec


U.S. ARMY
TRANSPORTATION CORPS
VZ-1AP

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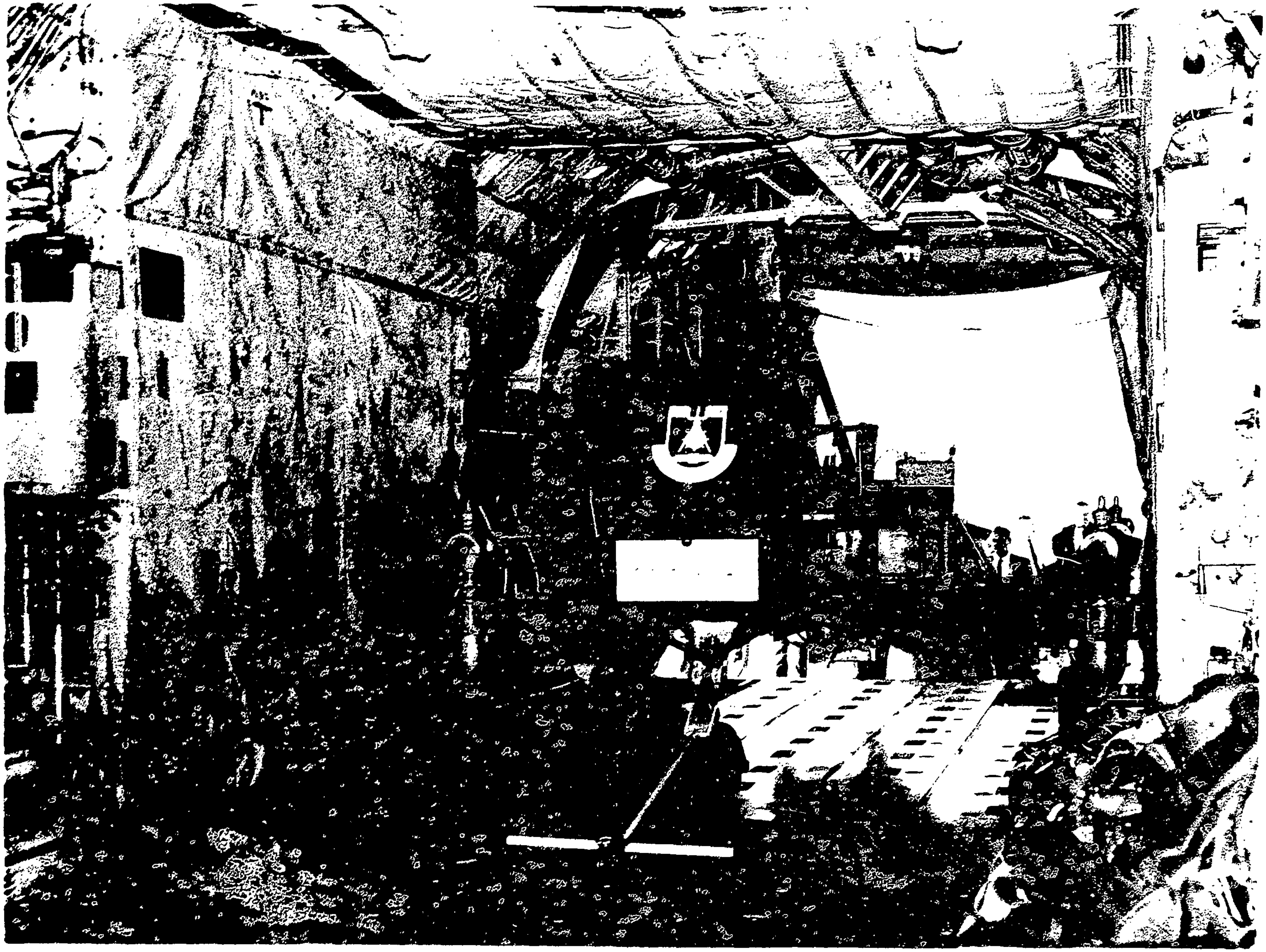

STRUCTURAL CONFIGURATION

The structural configuration was established around the basic concept of "put the members where the loads are". A series of widely divergent points of high loadings complicated the problem. These high loading points were:

Propeller thrust and pitching loads on corners of the basic structural box.
Propeller and gear box inertia loads under landing conditions.
Nose and main landing gear loads.
Engine support loads.
Tail wheel loads.
Cargo tie-down loads.
Main gear box support loads.

Preliminary design studies had pointed up the advantages of shear web and longeron construction as contrasted to a welded tubular truss scheme. The vibration associated with the proximity of the propeller tips to the structural webs had introduced the desirability of using shear webs of honeycomb material, rather than plain webs with a multitude of stiffening members. The usually troublesome problem of reliable attachments, particularly edge attachments to the honeycomb panels was solved by the use of custom-made panels featuring continuous shear webs, continuous edge doublers, and dimpled outer skins over the core of aluminum alloy honeycomb. This scheme is illustrated in Figure 18. These re-inforced honeycomb shear webs offered other advantages. such as:

a. Basic flatness and freedom from warpage, significant contributions in a limited tooling program.

b. Stiff panels to permit concentrated deck loadings as from aircrew, ground crew, and cargo, and

c. East of forming a natural cavity for the fuel cell.

Full length longerons of extruded 2024-T4 aluminum alloy were used to provide a continuity of structure from nose wheel to engine mount. Extrusions are also used as joining members at intersections of shear webs.

High strength 220-T4 aluminum alloy sand castings are used to carry principal loads into the basic box structure. Castings were used in lieu of forgings as a concession to cost and delivery schedules but at a sacrifice in vehicle payload. Castings were used at the following points:

At each corner of the basic box to pick-up propeller support arms.
At rear inside corners to serve as engine mount attach points.
On outside of box for main landing gear pivots and shock cord supports.
On center traverse web to support main gear box.

Attachments between shear panels and extrusions are AN 470DD6 rivets. Attachments through the panels and extrusions to pick-up the castings and other highly loaded items are AN series bolts or NAS series screws. Self locking nuts are used exclusively.

STRUCTURAL ANALYSIS

A complete structural analysis of the airframe was accomplished. The critical condition for bending and shear of the fuselage (as well as the main landing gear) was found to be the landing condition in which the main gear is first to contact the ground. A limit load factor of 3.5, with an ultimate factor of safety of 1.5, was used for this analysis.

Analysis of the control system was based upon the maximum loads which the pilot is assumed to be capable of applying to the stick, collective pitch lever and rudder pedals, in accordance with Specification MIL-S-5707.

The nose landing gear was checked for landing, towing and taxiing conditions, in accordance with ANC-2, "Ground Loads". The critical condition was found to be the three wheel landing conditions.

The structural analyses are a part of the Phase II Final Report, reference 32.

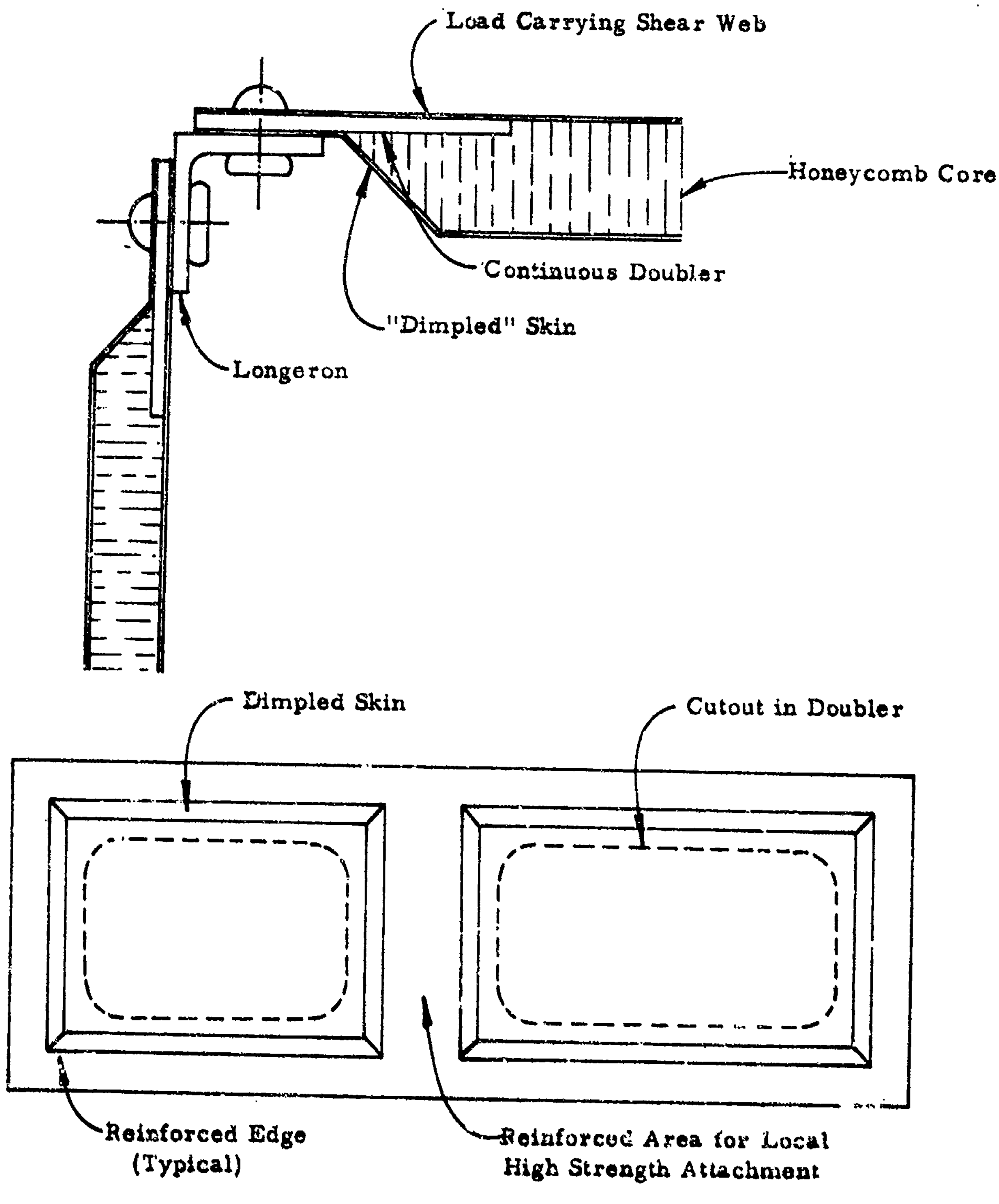

Figure 18 HONEYCOMB STRUCTURAL DETAILS

POWER TRAIN

Engine

The Aerial Platform is powered by a T-51 shaft turbine which develops approximately 400 shaft horsepower at take off rating. Performance characteristics of the engine are presented in Figure 19.

The engine delivers power to the Aerial Platform gear system through a 6000 RPM output shaft. Internal gearing is provided to reduce the turbine shaft speed of 34,800 RPM to the 6000 RPM output speed. The engine is equipped with an isochronous governor which maintains pre-selected engine speed.

Centrifugal Clutch

A centrifugal clutch is provided with the engine and permits the engine to start without carrying the inertia drag of the propellers, gears, and shafting. This clutch is designed to engage completely at approximately 25,000 RPM turbine speed. No slippage is expected at the operating speed of 34,800 RPM. With clutch engaged, the entire power train from engine to all propellers becomes a locked train with no independent speed control of any one element.

Input Coupling

A short universally mounted input shaft is installed between the centrifugal clutch and the central or main gear box. This shaft is coupled to the clutch and to the pinion shaft in the main gear box by means of spur gear type universal couplings capable of absorbing as much as ± 3° shaft misalignment. This requirement stems from the engine being shock mounted while the main gear box is rigidly attached to the airframe structure. This shaft is equipped with carbon face seals supported in metallic bellows to contain the clutch oil at the rear end of the input shaft, and to seal off the main gear box at the forward end of the input shaft.

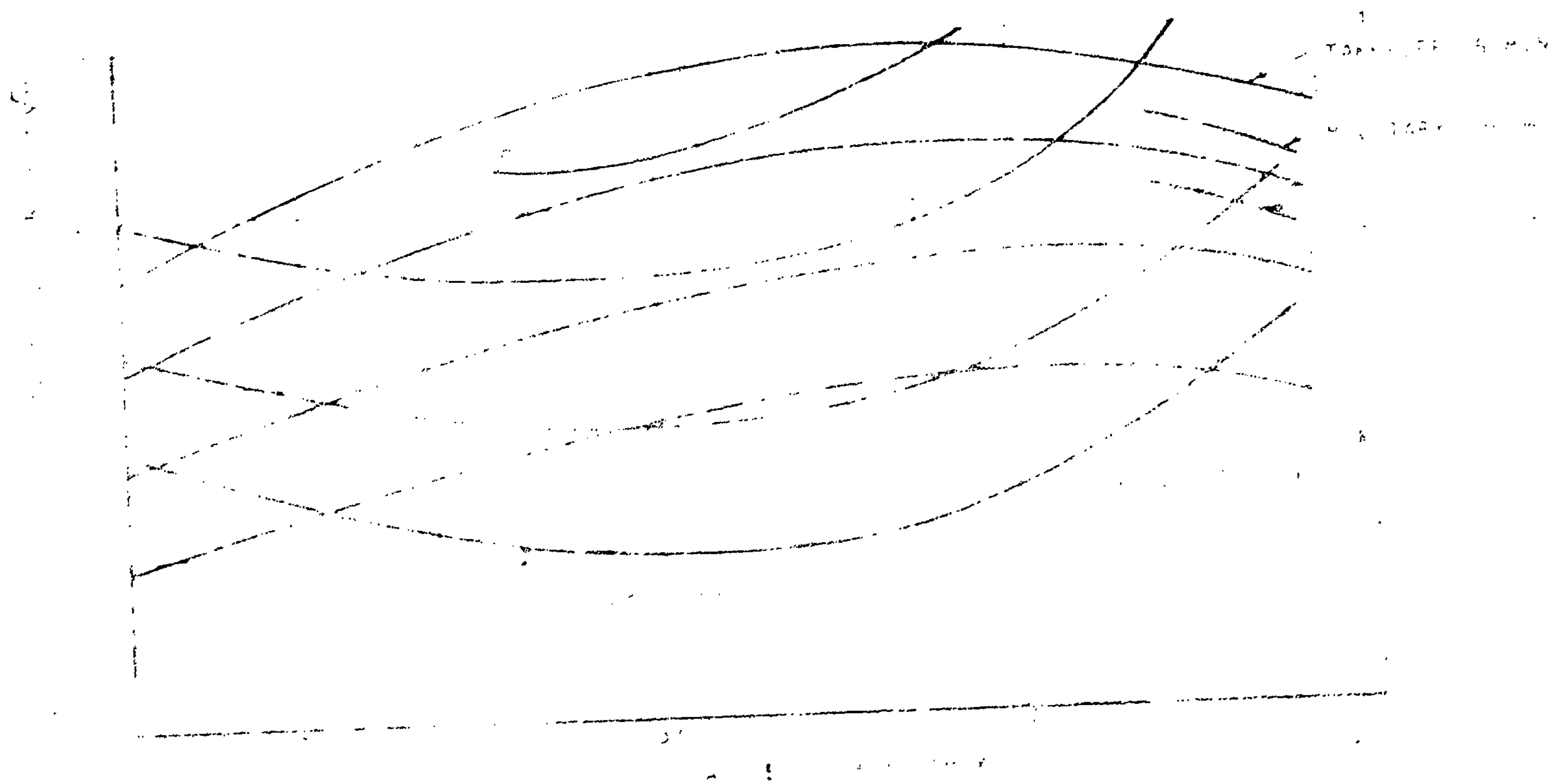

Figure 19 T-51 SHAFT TURBINE PERFORMANCE CHARACTERISTICS.

Main Gear Box

The main gear box accepts power from the engine at 6000 RPM and distributes it to the four propeller gear boxes through interconnecting drive shafts rotating at 3780 RPM. This speed reduction and the re-direction of the power is shown in Figure 20 The main gear box has a pinion shaft carrying two spiral bevel gears, each of which meshes with 2 mating gears which drive the intermediate shafts which lead diagonally to the four propeller gear boxes. See Figure 21. In the design of the transmission system careful consideration was given to fatigue. Wherever possible the shafts were designed to operate at stresses less than the endurance limit. This required the use of generous fillets and the minimization of stress raisers. Gears and bearings were designed for limited but adequate life (200 hours), in order to keep their size, weight, and housing weight to a minimum. In general, the housing design was controlled by the rigid support required by the gears, in order to maintain proper meshing under heavy tooth loads.

Principal features of the main gear box are ground spiral bevel gears to ensure smooth and vibration free operation, opposing layout of gears to balance radial and thrust loads, temperature probes on all bearings to provide warning of impending trouble, lubricant sprayed on all meshes and bearings, magnetic chip detector (not shown) fitted in the lower part of the oil sump, shims provided to facilitate precise meshing of the gear teeth.

Intermediate Drive Shafts.

Each propeller gear box receives its power from a tubular aluminum alloy shaft suspended between the central gear box and the propeller gear box. Each end of the shaft is fitted with spur gear type universal couplings to accept minor amounts of misalighment as would be associated with flight and/or landing loads. No intermediate or mid-span supports are used. Shaft proportions were chosen to stay safely away from an y shaft critical speeds. Lubrication for the universal couplings is provided as an integral part of the gear box lubrication system at each end of the shaft. Special attention to end connection design on this shaft provided a vernier-like means for orienting all propellers in the proper rotational position. This feature permitted broad freedom in machining of gear teeth in relation to splines and bolt patterns.

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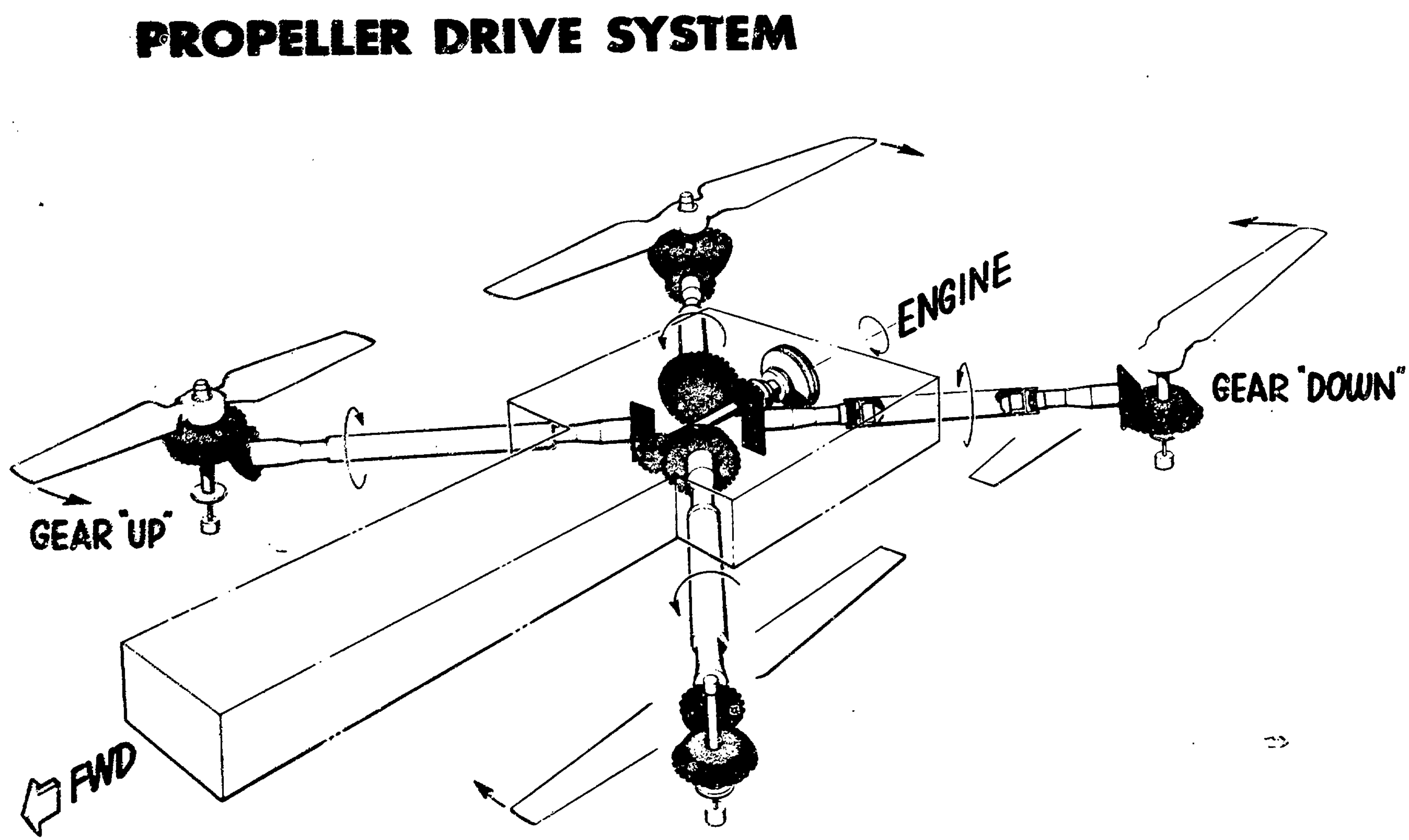

Figure 20 GENERAL ARRANGEMENT OF GEARS & SHAFTS

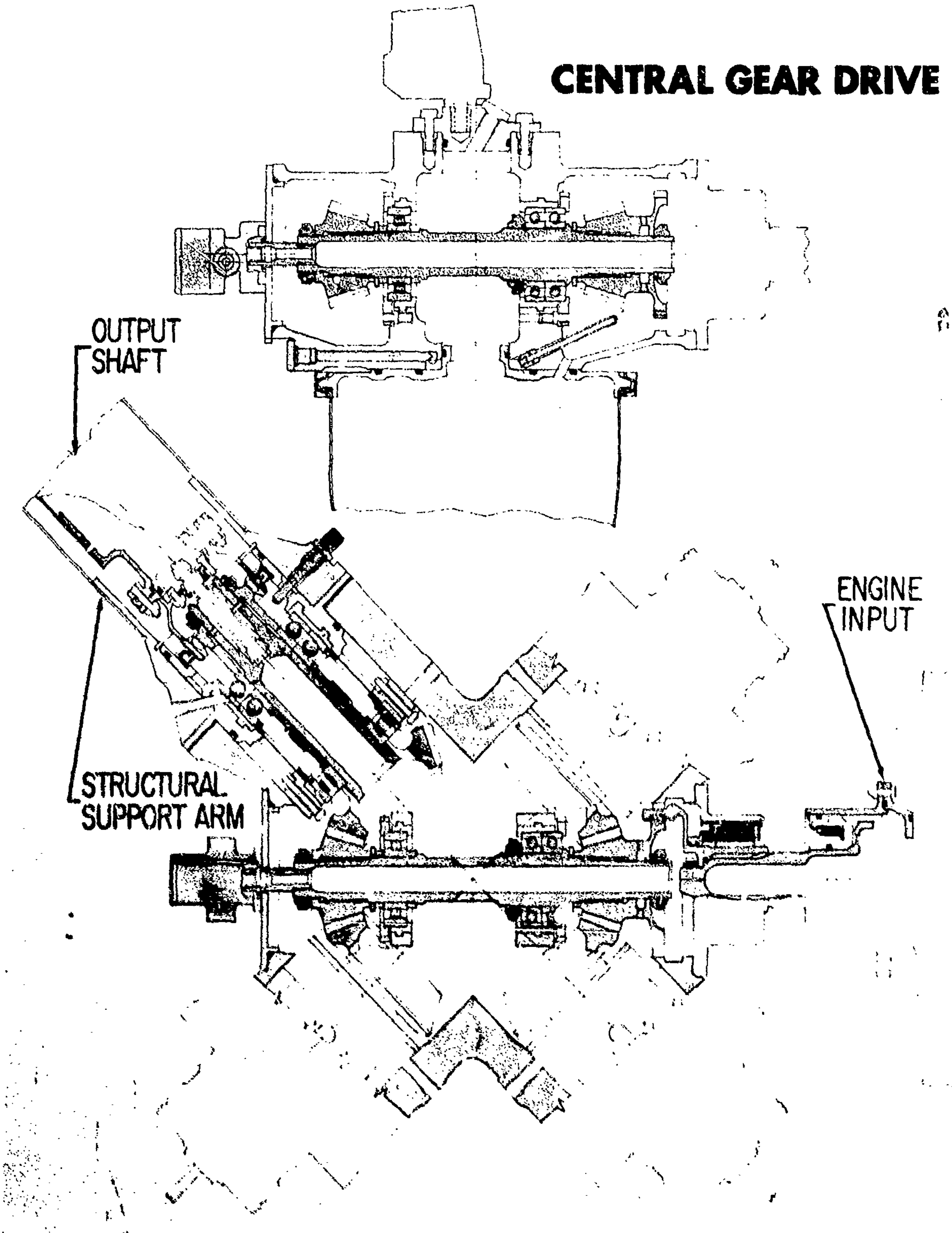

Figure 21 CUTAWAY EW OF MAIN GEAR BOX

Support Arms

Each propeller gear box is supported structurally by a support arm rigidly attached to the main gear box and to the basic platform structure. The support arms are fabricated of sheet aluminum alloy elements riveted to form a rectangular cross-section with semi-circular top. At each end is riveted a machined 356-T6 aluminum alloy casting. These castings provide the necessary precision fits to mate with the respective gear boxes. At approximately mid-span on each arm is riveted another 356-T6 aluminum alloy casting which mates with the corner fitting of the basic platform structure.

Propeller Gear Boxes

Each propeller gear box receives power from the main gear box and in turn delivers power to its rotating propeller. A speed reduction from 3780 RPM intermediate shaft speed to 2175 RPM propeller shaft speed is accomplished by means of a right angle spiral bevel gear drive. See Figure 22. The basic similarity of all four propeller gear boxes led to a design objective of making them interchangeable. Specific differences in requirements in regards directions of propeller rotation and "hands" of spiral gears precluded achieving this goal. Individual parts are, however, interchangeable in every possible instance.

Each propeller gear box consists of three major sub-assemblies mated to the parent housing. This permits bench assembly in advance and simplifies shimming for gear mesh adjustment at final assembly. Gear box housings (both main gear box and propeller gear boxes) are made from 356-T6 aluminum alloy sand castings with cast-in-place steel inserts to support the anti-friction bearings. The selection of this alloy was based on its excellent foundry characteristics, particularly in difficult sections.

The propeller gear box also houses the control system for the propeller hydraulic boost system. The hydraulic system consists of two gear type pumps, two check valves, two inlet filters, one relief valve and one reservoir. This hydraulic system delivers fluid under pressure to the propeller blade pitch changing mechanism and also supplies lubricating oil to the gear mesh points, gear couplings, and bearings. It is important to note that the hydraulic system of each propeller gear box is completely self-sufficient and is independent of all other gear boxes. In this way, otherwise lengthy, heavy, and vulnerable hydraulic tubing runs were eliminated.

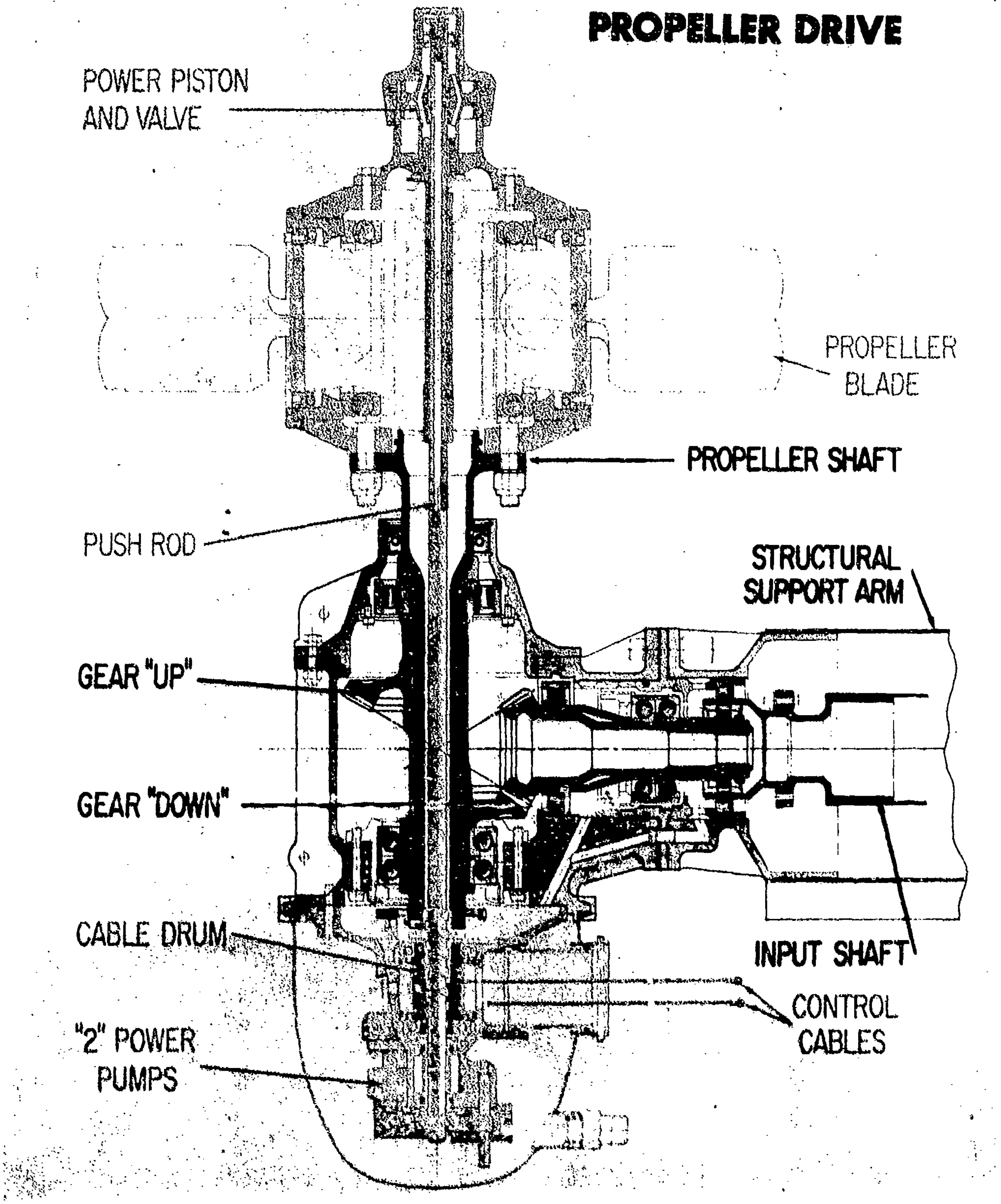

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Figure 22 CUTAWAY VIEW OF PROPELLER DRIVE

PROPELLERS

General Configuration

Two bladed propellers were a part of the basic vehicle configuration due to the requirement that vehicle fit into a USAF C-130 transport airplane. The use of three or more bladed propellers would have required removal of propellers for air transport.

Propeller Hubs

Basic similarities in size, power to be absorbed, and centrifugal forces between the Aerial Platform propellers and those of smaller aircraft led to the selection of a standard hub design with modified pitch change system. The propellers are bolted by typical flange mounting methods to the output shaft of each propeller gear box.

Propeller Blades

Propeller blades were designed by Curtiss-Wright Corporation, Santa Barbara Division. These blades are of laminated wood construction, covered with fibreglas, and protected from leading edge abrasion by a rubber strip placed approximately from mid-span to the tip of each blade.

Metal components to provide for blade retention, flapping freedom, etc., were designed and fabricated by Curtiss-Wright Corporation, Santa Barbara Division.

Aerodynamically, the propeller blades are both tapered and twisted following aircraft propeller design techniques rather than the constant chord, untwisted blades more typical of helicopters. Laminar flow sections (NASA 64_A series, 15% max. thickness) were selected for maximum propeller performance.

The aerodynamic design of the propellers is contained in reference 32.

Propeller Pitch Control

The original goal of unboosted manual control of the propeller blades (based on desired simplicity) was found to be impractical due to the large blade control moment requirements. Centrifugal forces on the blades, coupled with aerodynamic forces, dictated some form of power assist for the pilot. Ingenious systems of mechanically accomplishing this boost were considered, but discarded in favor of time-proven systems of hydraulic servo control.

The one-way hydraulic control normally associated with airplane propellers was discarded in favor of a balanced servo system which would assure proper response.

The propeller boost unit was designed, developed, and manufactured to basic specifications established by Curtiss-Wright, Santa Barbara Division. Fluid supply for the hydraulically operated blade pitch changing mechanism is delivered to the propeller through a hollow valve stem which does not rotate. The axial position of this stem is controlled by the pilot through the cable system which terminates in a lead-screw mechanism mounted beneath the propeller gear box housing. This stem and its corresponding piston form a balanced servo system wherein the piston (controlling the blade pitch angle) follows precisely the axial movements of the valve stem. See Figure 23.

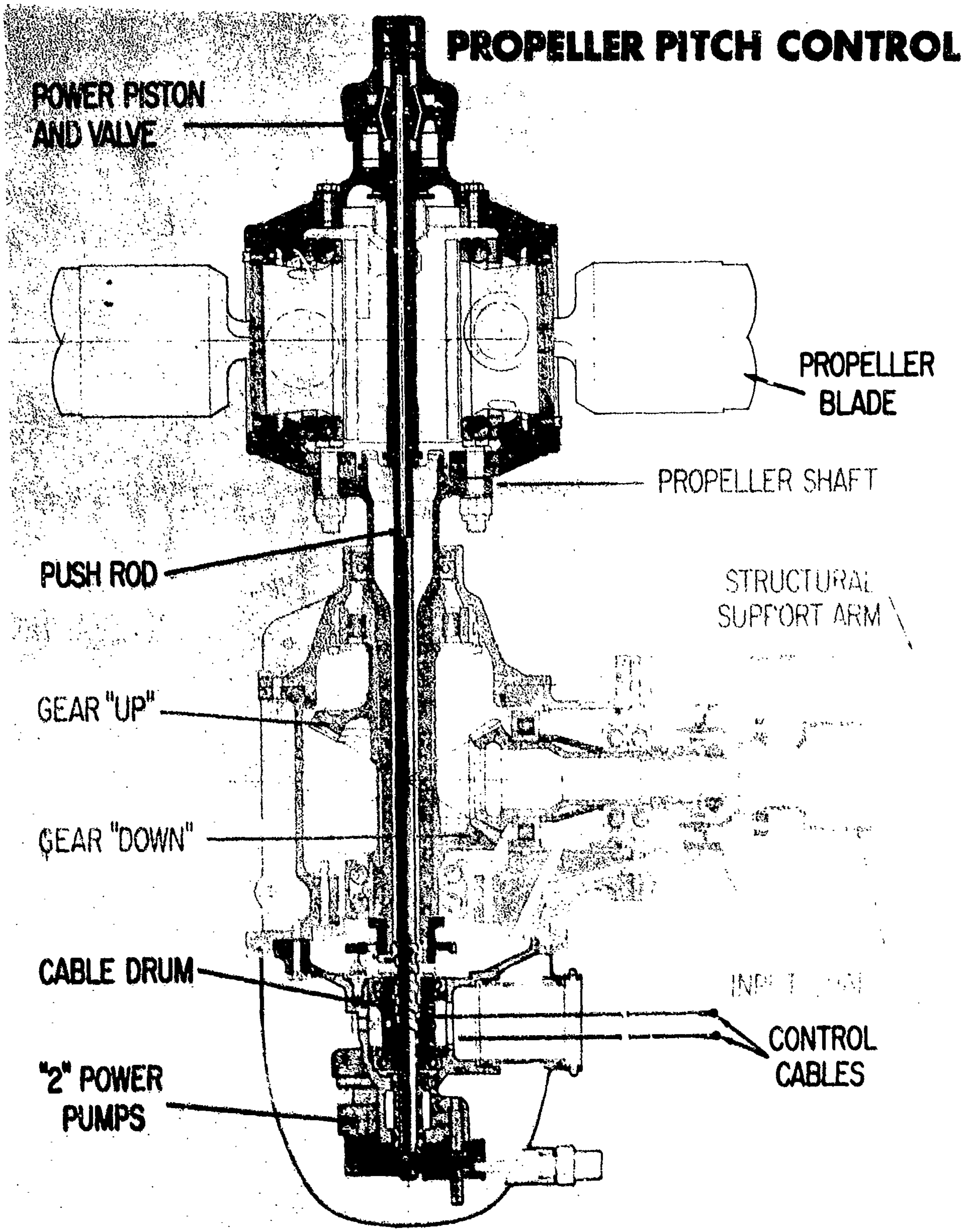

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Figure 23 CUTAWAY VIEW OF PROPELLER PITCH CONTROL

LANDING GEAR

General Configuration

The general arrangement of the tricycle type landing gear was established in the preliminary design phase to provide greatest ground stability and protection for the propellers commensurate with weight, simplicity, etc. A small fixed tail wheel was added under the engine tail pipe for protection in case of nose high landings and is not considered to be a working part of the normal landing gear.

Main Landing Gear

The main landing gear consists of two symmetrically opposite tubular steel trusses attached to the basic platform structure by means of a large 220-T4 aluminum alloy sand casting machined to form the principal fulcrum. The wheel and brake assembly (5:00-4) is mounted to the lower (rearward) end of the truss while the upper (forward) end carries several loops of shock cord. These shock cord loops are also attached to the basic platform structure by means of a specially designed fitting machined from a 220-T4 aluminum alloy sand casting. Ground contact forces against the tire are reacted through the wheel, the axle, the truss, the main fulcrum and through the shock cord loops. Reduction of the ground contact forces is accomplished by stretching these shock cord loops while the wheel undergoes a nearly vertical stroke, similar to that experienced in more sophisticated systems utilizing oleo shock absorbers. Each main landing gear wheel is fitted with an expander tube brake to permit braking for maneuvering and parking and for resisting the residual thrust of the turbine exhaust.

Nose Landing Gear

The nose landing gear also consists of tubular steel framework elements. The wheel is full swivelling to provide freedom for ground mobility and to reduce loads associated with side drift landings or ground maneuvers. A centering cam is provided to align the nose wheel with the fore and aft axis of the vehicle. The nose gear is fitted with the same size wheel and tire as the main gear (5:00-4) though without brakes. Provision is made on the nose landing gear for attachment of a tow bar to facilitate ground handling.

Shock absorption on the nose landing gear is provided by the same rubber shock cord scheme as described above for the main landing gear. An advantage not readily apparent, is that the extended and static load positions of this landing gear are essentially the same. For an oleo type gear the static position would more nearly correspond to the fully compressed position. On the VZ-7AP with an equivalent stroke of 8 inches, this would mean that the propellers would be 6 or 7 inches closer to the ground in the static case. The safety advantages of having the propellers higher above the static ground line are obvious.

FLIGHT CONTROL SYSTEM

Basic System (See Figure 24)

The flight control system of the Aerial Platform is designed to provide attitude control about the three principal axes, plus providing altitude control. These functions are accomplished as follows:

Pitch attitude is controlled by a fore and aft motion of the main (differential) control stick, operated by the pilot's right hand. The pitch control moment is produced by simultaneously adding blade pitch to the two forward propellers while reducing blade pitch on the two rear propellers or vice versa.

Roll attitude is controlled by lateral motion of this same differential control stick. The roll moment is produced by adding blade pitch to the right hand propellers while reducing blade pitch on the left or vice versa.

Yaw attitude is controlled by means of rudder pedal motion by the pilot's feet. The yaw moment is produced by a series of four (4) vanes mounted beneath the rear propellers to deflect a portion of the slipstream.

Altitude is controlled by means of a collective pitch lever operated by the pilot's left hand. Raising the lever (pulling back) simultaneously increases the blade pitch on all four propellers whereas lowering the lever (moving it forward) reduces blade pitch.

Control Mixer

The formidable task of accumulating both collective and differential stick position inputs and correlating them into a family of coordinated blade pitches falls to a device referred to as the "mixer". This is a purely mechanical device consisting of push rods, bell cranks and sliding pivot blocks. This mechanism is shown in Figure 25. The outputs from this mixer are angular motions of cable quadrants mounted beneath the mixer, i.e., on the underside of the platform structure. These quadrants can be seen in Figure 26.

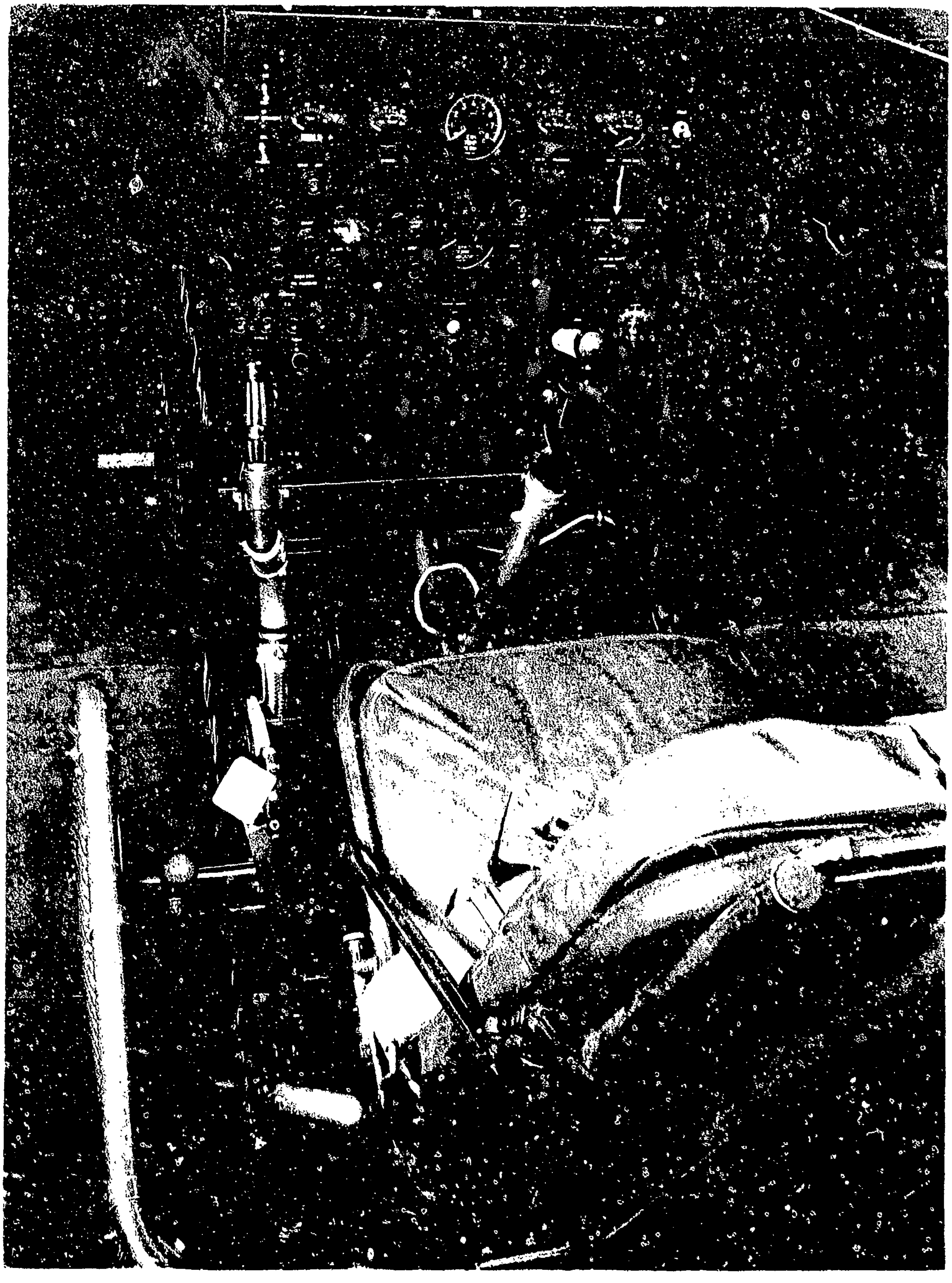

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Figure 24 COCKPIT AND CONTROLS

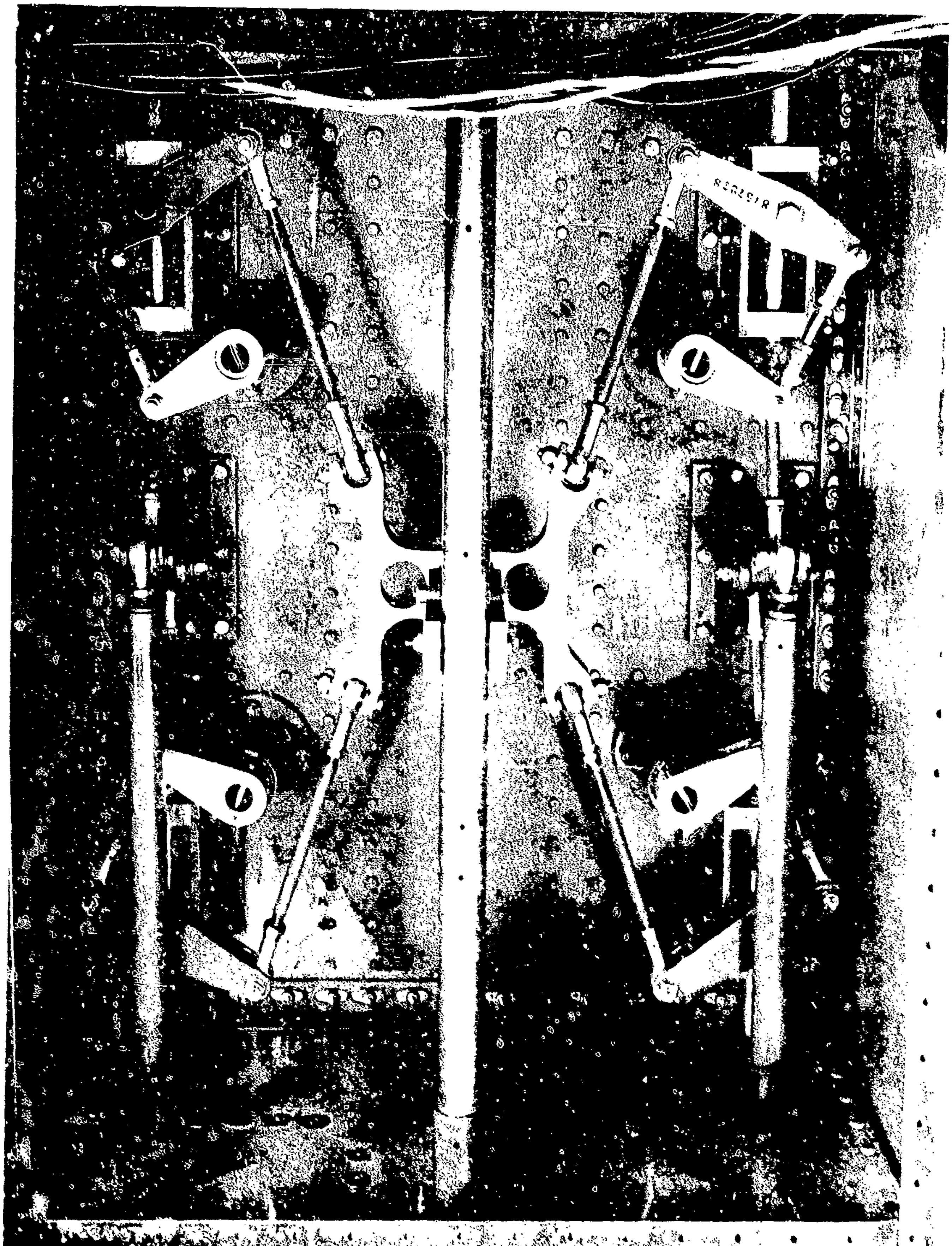

Figure 25 CONTROL MIXER INSTALLATION

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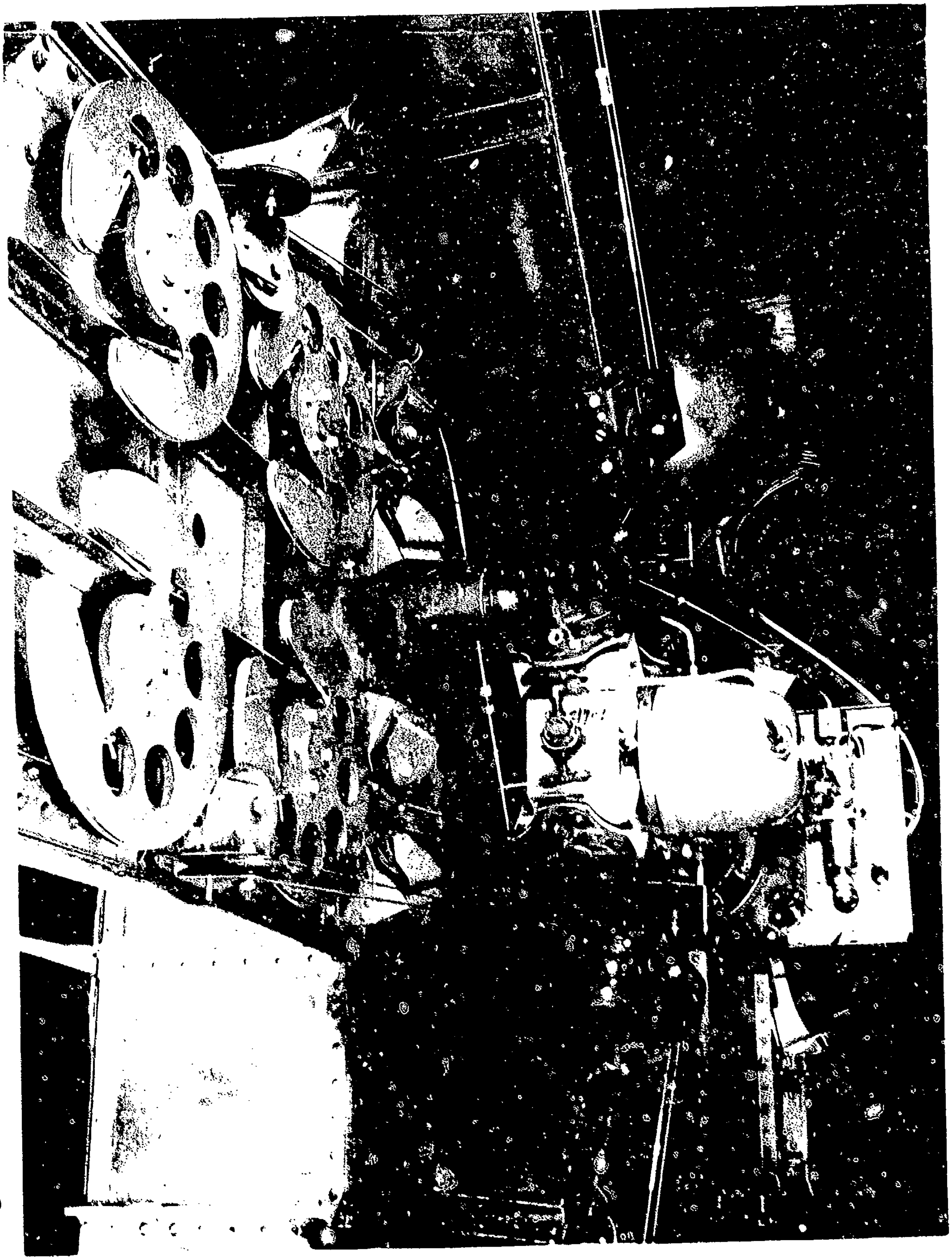

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Cable System

The output motions of the mixer, i.e., quadrant motions, are transmitted to the individual propeller gear boxes for propeller blade pitch control through a system of steel cables. This use of cables rather than push-pull tubes or torque tubes provided necessary design freedom in choice of routings and virtually eliminated the potentially adverse effects which structural deflections would have on push-pull or torque tube systems. The outboard terminal point for each propeller control cable system is the lead screw mechanism which operates the hydraulic boost valve stem. This is described in the section titled "Propeller Pitch Control".

Dual Controls

The VZ-7AP Aerial Platform was designed as a two place vehicle and as such provisions for dual controls were incorporated into the design. The prototype vehicle was rigged for operation from the front cockpit only. Rudimentary control elements are installed in the rear cockpit so that a complete system of dual controls could be readily installed.

Control Forces

The control system includes antifriction bearings throughout in order to limit friction to acceptable values. Springs are used in the pitch, roll and yaw control systems to provide an artificial force feel gradients.

Measured values of control forces are as follows:

Roll (right or left)	(Breakaway	24 ounces
	(Hard over	48 ounces
Pitch - forward	(Breakaway	42 ounces
	(Full forward	54 ounces
Pitch - aft	(Breakaway	42 ounces
	(Full aft	64 ounces
Collective Pitch (increasing or decreasing)	5 to 6 lbs maximum	

ENGINE CONTROL SYSTEMS

Engine Controls - Manual

The pilot has two normal engine controls and one emergency control, none of which require attention from the pilot during flight.

Fuel Shut-Off Valve. This controls the flow of fuel other than that required for starting and idling.

Engine Speed Governor Control. This lever permits the pilot to set the engine speed at a predetermined point. Thereafter, regardless of load changes, the govenor will control fuel flow to maintain the pre-selected speed.

Emergency Fuel Shut-Off Valve. This is provided to mechanically shut-off all fuel flow to the engine. This control is used only in the event that the normal start and idle fuel shut-off valve (electrically controlled by the ignition switch) fails to function.

All of the above engine controls are operated by push-pull controls operating in rigid stainless steel tubing conduit. The control levers and/or handles can be seen in Figure 24.

Engine Controls - Electric

The Aerial Platform is equipped with the automatic starting sequence control box furnished with the engine. Operation of the starting and stopping system is reduced to a simple on-off function, with further provisions for venting the engine, i.e., motoring it without fuel or ignition systems operating.

FUEL SYSTEM

General Description

The fuel system in the Aerial Platform is designed to provide filtered JP-4 fuel to the engine in sufficient quantity and at suitable pressure to permit maximum power output from the engine. A quantity of 200 pounds of fuel is carried. This corresponds to 30 minutes endurance at maximum power. The fuel system consists of the following elements:

Fuel Cell

The fuel cell is a flexible nylon and synthetic rubber bladder supported inside a structural cavity in the platform structure. This type of cell was chosen due to its light weight and inherent durability. The fuel cell and cavity have been contoured to provide an inverted pyramid drain-towards-center-fuel sump. Fuel is trapped within a center baffle area to assure uninterrupted fuel flow when the fuel level is low and the vehicle is not in a level attitude.

Vent System

Two fuel vents are provided. The left hand one by means of a line tapped into the filler neck and the right hand one by a short tube routed forward and upwards. The vent system provides for minimum fuel leakage during maneuvers and maximum venting of vapors during re-fueling. Control of the maximum fuel level is accomplished by closing the right hand vent during re-fueling operations.

Water Sump

A sump for collecting water condensation and foreign matter is built into the bottom plate of the fuel cell in such location as to preclude contamination of the fuel. A military type self-locking drain cock is provided in this sump. This sump can be seen immediately behind the control quadrants in Figure 26.

Shut-Off Valves

A tank drain shut-off valve is provided for convenience and safety in servicing the fuel system downstream from the tank. This valve is safety wired in the "open" position for flight. A master fuel shut-off valve located at the engine and operated manually from the cockpit affords safety by complete shut-off of all fuel to the engine. This control is described under Engine Controls.

Filters

All fuel delivered to the engine passes through a micronic filter to assure cleanliness of fuel. In addition, another filter, furnished with the engine, further filters fuel to the starting fuel nozzles.

Pumps

Fuel is delivered under pressure to the engine by an engine driven fuel boost pump mounted on the accessory section of the engine. Fuel is gravity fed to this pump from the fuel tank. Fuel discharged from this pump is routed through the main fuel filter and thence into the engine. A second pump, engine driven, and internally mounted as part of the engine's own fuel system accepts flow from the above described boost pump and in turn delivers it to the engine speed governing unit. A third pump, electrically driven, and also furnished with the engine, supplies fuel under pressure during the starting sequence only. Control of this pump is one of the functions of the automatic starting control system on the engine.

ENGINE OIL SYSTEM

General Description Oil System

The engine oil system of the Aerial Platform handles oil for all engine lubrication requirements. It is completely independent of engine clutch oil and of the various propeller drive system gear boxes. The engine oil system consists of the following units:

Reservoir

An aluminum alloy container of spherical shape, mounted beneath the platform structure serves as the reservoir. It has a 2 gallon capacity. It contains an air separation screen, drain plug, temperature probe, filler neck, vent, and connections for oil inlets and outlets. This reservoir by virtue of its aluminum alloy construction also serves as a radiator for some of the excess heat picked up by the oil. The reservoir can be seen in Figure 26.

Engine Oil Pumps

The engine lubricating oil is circulated and scavenged by a series of pumps built into the engine. No external pumps are used.

Engine Oil Filter

The engine lubricating oil is filtered continuously by a filter furnished as an engine-mounted accessory. No external filtering is provided.

Oil Cooling Coils

Cooling of the engine lubricating oil is accomplished by circulating the oil through a simple coil of aluminum alloy tubing mounted beneath the airframe structure.

ELECTRICAL SYSTEM

Power Circuits

The primary source of power is an AN3154-1A 24 Volt DC lead acid storage cell. Its primary purpose is to energize the starter-generator during engine starting. After the engine has started, the start winding is removed from the circuit and the generator supplies power to the circuits through the voltage and current regulator. The regulator is adjusted for system voltage and will deliver 25 amps maximum. System voltage and current is displayed on the instrument panel on a combination ammeter and push-to-read voltmeter.

The power requirements for starting the engine are easily met by the above battery. Therefore, no provision for use of external power supply was made.

Power is distributed to the instrument circuit and the starting system circuit through respective circuit breakers. After the engine is brought to idle speed, the loss of electrical power by tripping a breaker on either or both circuits will not shut off the engine. If electrical power cannot be restored due to some malfunction, the engine can be turned off by operating the manual fuel shut off valve from the cockpit.

Instruments

Remote indicating electrical instruments are mounted conveniently in the instrument panel for easy read-out by the pilot. Engine instruments include tachometer, exhaust gas temperature, engine oil temperature, and engine oil pressure. Oil temperature gages are provided for the main gear box and with a selector switch for the four (4) propeller gear boxes. Electrical power is displayed on a combination ammeter and push-to-read voltmeter.

The intended test missions involved no requirement for flight instruments.

Warning Systems (See Figure 24)

A number of warning lights actuated by relays or switches are displayed on the panel. Four lights (mounted adjacent to the tachometer) labeled "starter", "stop", "power limit", and "fuel valve" are associated with the automatic starting system and serve to warn the pilot of engine starting or stopping malfunctions.

A family of four (4) lights is provided to warn the pilot of loss of propeller pitch control hydraulic power on any of the four (4) propeller gear boxes. A fifth light within this grouping would give warning of loss of lubricating oil pressure on the main gear box.

Another group of four (4) warning lights is provided to indicate sub-standard lubrication oil pressure at each of the four (4) propeller gear boxes.

A single chip detector light actuates when a chip in any gear box completes this warning light circuit.

WEIGHT & BALANCE SUMMARY

Discussion - Weight

In the original concept of the Aerial Platform an ultimate vehicle was envisioned which would retain the desirable characteristics of the military land based jeep plus possessing the ability to rise vertically, to hover, and fly forward at reasonable speeds. Toward these ends a payload requirement of 1000 lbs (including the crew) was proposed.

The research test bed vehicle provided for in Contract DA-44-177-TC-397 was not intended to fulfill all the desired objectives of this forecast ultimate vehicle. Therefore, a more modest payload goal was essential. This was emphasised by the obvious requirements for rugged construction to withstand unpredictable flight and landing loads and for suitably conservative ratings on the power train to assure reliability without extensive development and testing.

As the detail design of the vehicle progressed, a more firm picture of weight distribution requirements was evolved. The original gross weight of 1850 lbs was raised as more data was accumulated on the weight of engine, gear drive system, propellers, structure, landing gear, control system, etc. A top limit gross weight of 2400 lbs was established as a reasonable goal within which to develop the design and which appeared well within the thrust potential from the engine-propeller combination. Actual weights, measured thrusts and successful flight testing have since proven this 2400 lbs goal to have been a realistic value.

Weight Breakdown

Drive System	680.	
Engine, Mount, Tail Pipe, Tail Wheel Structure	380.	
Structure	248.	
Landing Gear	132.	
Fuel and Oil Systems	35.	
Controls	145.	
Battery and Regulator and Wiring (excess classed as payload)	37.	
Seat Installation (Turnover excess classed as payload)	18.	
Instrument Panel and Wiring	28.	
Yaw Modification	50.	
Total	1753.	
Weight Empty:		1753.
Useful Load & Payload:		
Oil	40.	
Fuel	210.	
Turnover Structure Excess	25.	
Battery System Excess	13.	
Pilot	180.	
Instrumentation or Alternate Loads	179.	
Total	647.	647.
Design Gross Weight		2400. lbs.

Drive System Weight Breakdown

4 Prop Gear Boxes @ 47.0	188.0 lbs
4 Support Arms @ 20.75	83.0
1 Main Gear Box @ 63.0	63.0
4 Drive Shafts @ 4.0	16.0
4 Propellers @ 75.5	302.0
8 Drive Shaft End Couplings	9.5
Miscellaneous (Hardware, etc)	19.5
Total	680.0 lbs

Battery System "Excess"

The battery originally used on the prototype vehicle is a large capacity lead-acid battery, which with regulator and mounting box weighs 72 lbs. This choice was predicated on unusually severe power drains antitipated in a program involving many engine starts, plus supporting the stability augmentor equipment and normal aircraft electrical functions. This choice was further supported by considerations of cost, availability, and simplicity of maintenance of the battery. During the weight reduction program this AN3151-2 battery was replaced with a lighter AN3154-1 for a weight savings of 22 lbs. See Modifications, page 123. Further weight savings could be realised by use of a still lighter battery, by redesign of the battery mount and by selection of a light weight voltage regulator. Therefore, a battery system "excess" of 13 lbs is listed as payload in the Weight and Balance Summary.

Overturn Structure "Excess"

The original seat support and turnover structure provided for pilot safety reasons on the prototype research vehicle represents a much larger weight investment than would be justified on a more completely developed vehicle. During the above noted weight reduction program a lighter weight overturn structure was fabricated, saving 26 lbs. Further weight savings could be realized by redesign of the overturn structure to a simple "roll-bar" as part of the seat. Complete removal of the overturn structure might be in order on operational configurations, i.e., variations in vehicle configurations to suit specific end uses might provide reasonable crew protection without the specifically separate structure used on the prototype. Therefore, an overturn structure "excess" of 25 lbs is listed as payload in the Weight and Balance Summary.

Payload Potential

For the research vehicle the useful load and payload rating is 647 lbs for a design gross weight of 2400 lbs as listed above.

This research test bed vehicle has been flown at a working gross weight of 2545 lbs and an overload gross weight of 2900 lbs.

It is therefore well within reason to expect that a modified test bed could demonstrate a combined useful load and payload of 770 lbs.

Discussion - Center of Gravity Location

Longitudinal

For a four rotor vehicle designed for maximum lift and for hovering flight only, the optimum logitudinal location for the center of gravity would be at the geometric center of the pattern of the four rotors. However, when the vehicle moves forwar[illegible]stantial pitching-up moment is evidenced. This can be offset by increasing lift on the rear rotors while decreasing lift on the forward rotors. Examination of the magnitude of these moments disclosed that a compromise of the optimum hovering C. G. location would be in order. The center of gravity for the prototype vehicle was established for design purposes at 10-12 inches ahead of the geometric center of the pattern of the rotors. The range of longitudinal C. G. locations actually used in our flight test program is shown in Table 6, page 127.

One of the characteristic advantages of the four rotor Aerial Platform over a single or dual rotor helicopter is the relative insensitivity of C. G. changes. This was borne out in the test program where logitudinal C. G. shifts of 9.25% of rotor radius were accepted with no problems.

Vertical

The vertical location of the C. G. was virtually fixed from the beginning, due to the low silhouette and the relatively concentrated vertical arrangement of masses. The calculated vertical C. G. location is 3[illegible].2 inches above the static ground line. This corresponds to 6.8 inches below the loading deck or 2.4 inches below the plane of the propellers. Flight test variations from this normal vertical location are shown in Table 6, page 127. Vehicle handling characteristics were not changed detectably by these shifts in vertical C. G. location.

Moment of Inertia

The moment of inertia of the vehicle was determined by a swing test on calibrated shock cord springs. Data from that test and subsequent calculations follows:

	As Swung:	At the then current gross weight:
Weight	1936. lbs	2381. lbs
C. G.	Sta. 193.8	Sta. 186.7
"I"-Pitch	820 Slug-ft^2	1055 Slug-ft^2
"I"-Roll	410 Slug-ft^2	467 Slug-ft^2
"I"-Yaw	1190 Slug-ft^2	1476 Slug-ft^2

4. PRELIMINARY TESTING

Introduction

Prior to the free flight testing of the VZ-7AP Aerial Platform, a number of tests of various types were conducted to verify the performance of certain components of the total vehicle. These tests included propeller tests, gear box tests, vibration measurements, total thrust measurements, and tethered flight testing.

These preliminary tests will be the subject of this section of this report.

Propeller Tests on Truck

Preliminary tests were run on one full-scale propeller on a test rig using a V-belt drive from an automobile engine. The entire test rig was mounted on a flat bed truck and operated first with the truck stationary to simulate hovering flight conditions and then with the truck moving to simulate forward flight. Strain gage instrumentation was utilized to measure torque and thrust. At a representative blade pitch angle, a thrust of 600 lbs. required 100 shaft H. P. These values are gross values and do not include any correction for flow disturbances around the truck and test rig. A 30% overspeed test was performed to evaluate the structural integrity of the propeller. (30% overspeed equals 69% overload.) No evidence of failure or permanent set was detected.

Propeller Proof Tests

Prior to the installation of propeller assemblies to their respective gear box, a propeller proof test was conducted. The propeller was mounted on a 50 H. P. hydraulic motor which in turn was secured to a large steel test fixture. Each propeller assembly was rotated at 12% to 15% overspeed. (25% to 32% overload.) Inspection of each propeller assembly indicated that no structural failure or permanent set occurred during these runs.

Propeller Gear Box Tests

Upon completion of the gear box assembly each unit was tested in a dynamometer and then inspected for gear mesh. Adjustments were made and then the complete gear box and propeller assembly were tested. Propeller response, control cable travel, pump pressure, bearing temperature, and RPM were monitored.

The first of the four propeller gear boxes was subjected to a series of test runs totalling 4 hours for detection of rapid wear characteristics as well as performance. The other 3 propeller gear boxes were tested similarly but for only 1/2 hour each. Gear box and propeller control performance was found to be entirely satisfactory.

Main Gear Box Tests

The main drive gear box was tested for gear pattern in a dynamometer. Inspection and adjustment was followed by 4 hours testing at various speeds and torque loadings representing all operating conditions, including exaggerated angular offset of the input shaft, and unsymmetrical loading of the output shafts. Performance was found to be entirely satisfactory.

Ground Tied-Down Tests, Vibration

Tied-down runs of the entire vehicle were made to assess vibration characteristics. Potentially harmful vibrations of large amplitude were experienced at approximately 10 cycles per second in the propeller support arms. Vehicle modifications were undertaken to limit the amplitude to acceptable levels. The tubular struts between the propeller gear boxes and between the platform structure and the forward propeller gear boxes were added during this test phase. (Compare Figure 16 with Figure 28.)

The propeller blades had originally been fabricated with limited freedom in the drag direction as well as in the flapping direction. Close study and consultation with NASA, Langley Research Center, established these drag hinges as the major offenders in the vibration encountered. The drag hinges were locked out concurrently with the installation of the tubular struts noted above.

Subsequent testing showed no harmful vibration.

Thrust Measuring Tests

Thrust measuring tests were made with the vehicle tied-down at normal ground level and with the vehicle elevated 4 feet above normal ground level. (See Figures 27 and 28.) Comparison of the results permitted assessment of ground effect on the thrust or lift of the propellers.

For the ground tests and the elevated frame tests, the platform was tied down approximately under each propeller gear box. Between the ground attach ring and each of the four platform tie-down points, a load ring was connected to the link system to measure thrust. The load ring sensing system consisted of four SR-4 strain gages connected in a full bridge (all four gages active and temperature compensated). Each end of the load ring was pin-ended connected so as to eliminate the actions of side forces and eccentric loading. The load ring, therefore, indicated only the axial forces through the tie-down link.

The load rings were calibrated with a Tinious-Olsen 20,000 pound universal testing machine to a load of 1500 pounds. The calibration signals were compared with the signal from a 0.1% calibration resistor built into each load ring circuit. Signals from the calibration resistor served as the calibration mark made on every record before and after every tie-down test for thrust measurement.

Test runs were made to determine thrust variations through a full range of collective pitch, and full ranges of differential roll and differential pitch controls. Thrust data, collective stick position, differential stick position and propeller blade angles were recorded simultaneously. Thrust vs blade angle is presented in Figure 29 . The two curves clearly show the increased lift benefit from ground effect.

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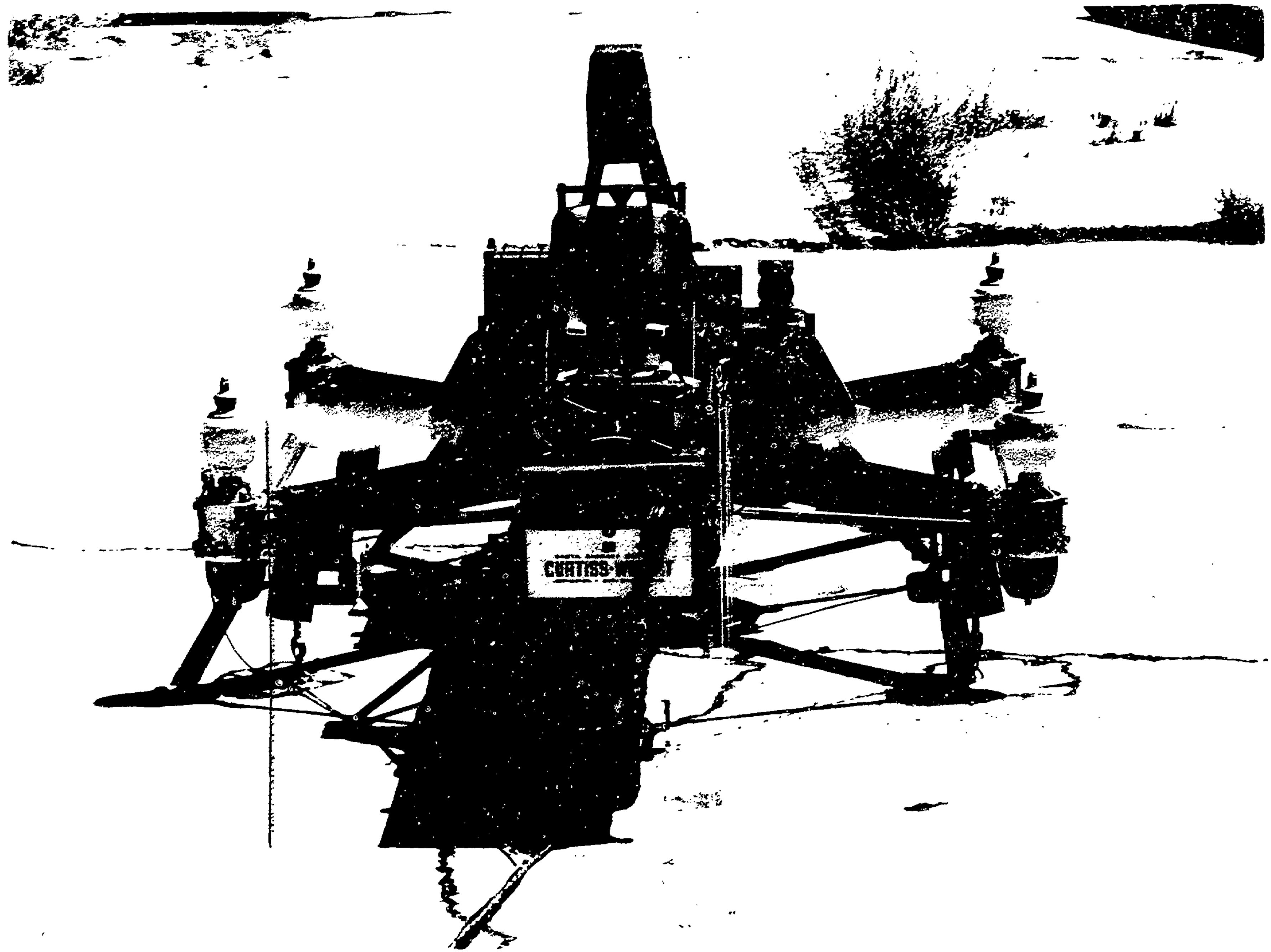

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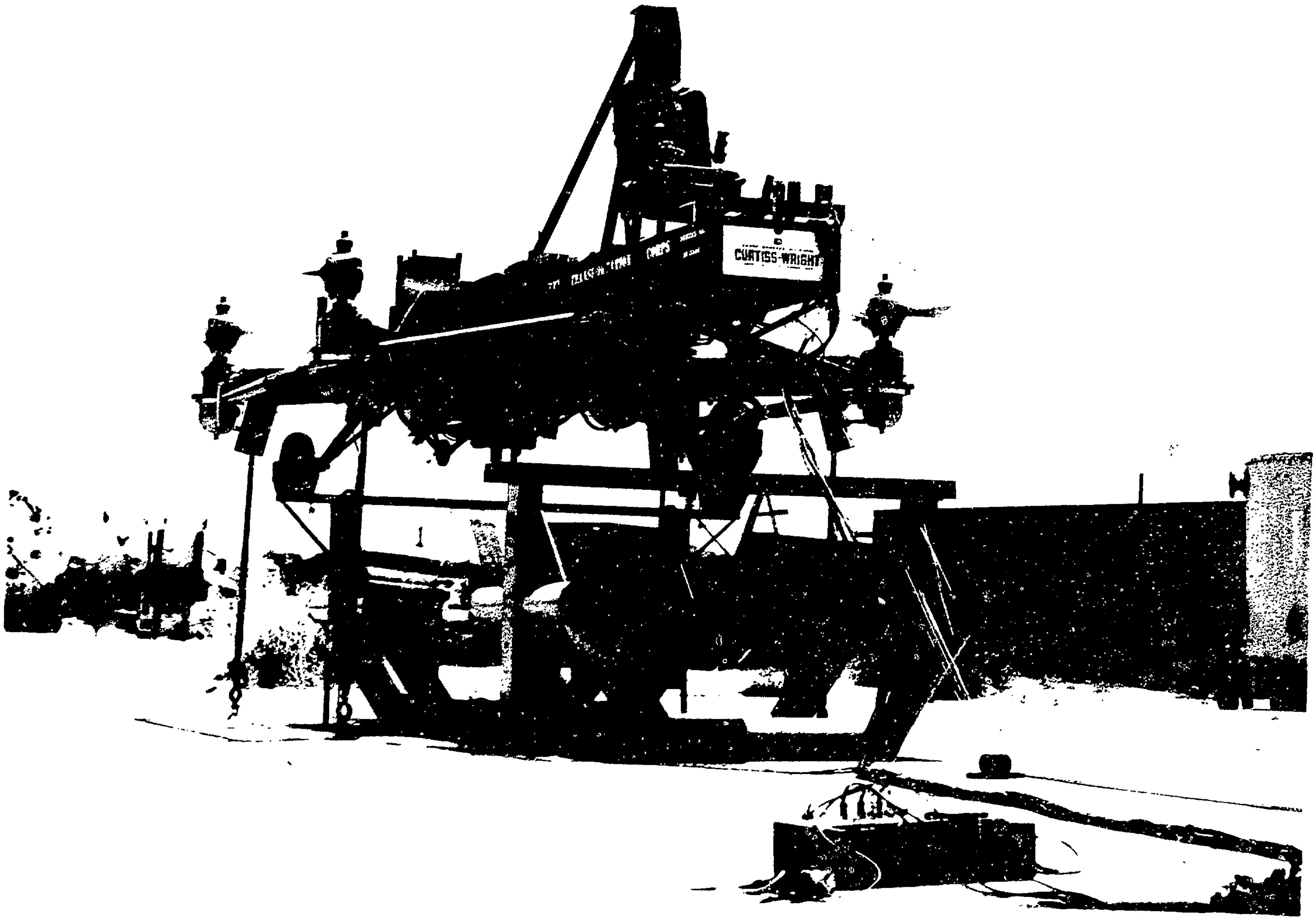

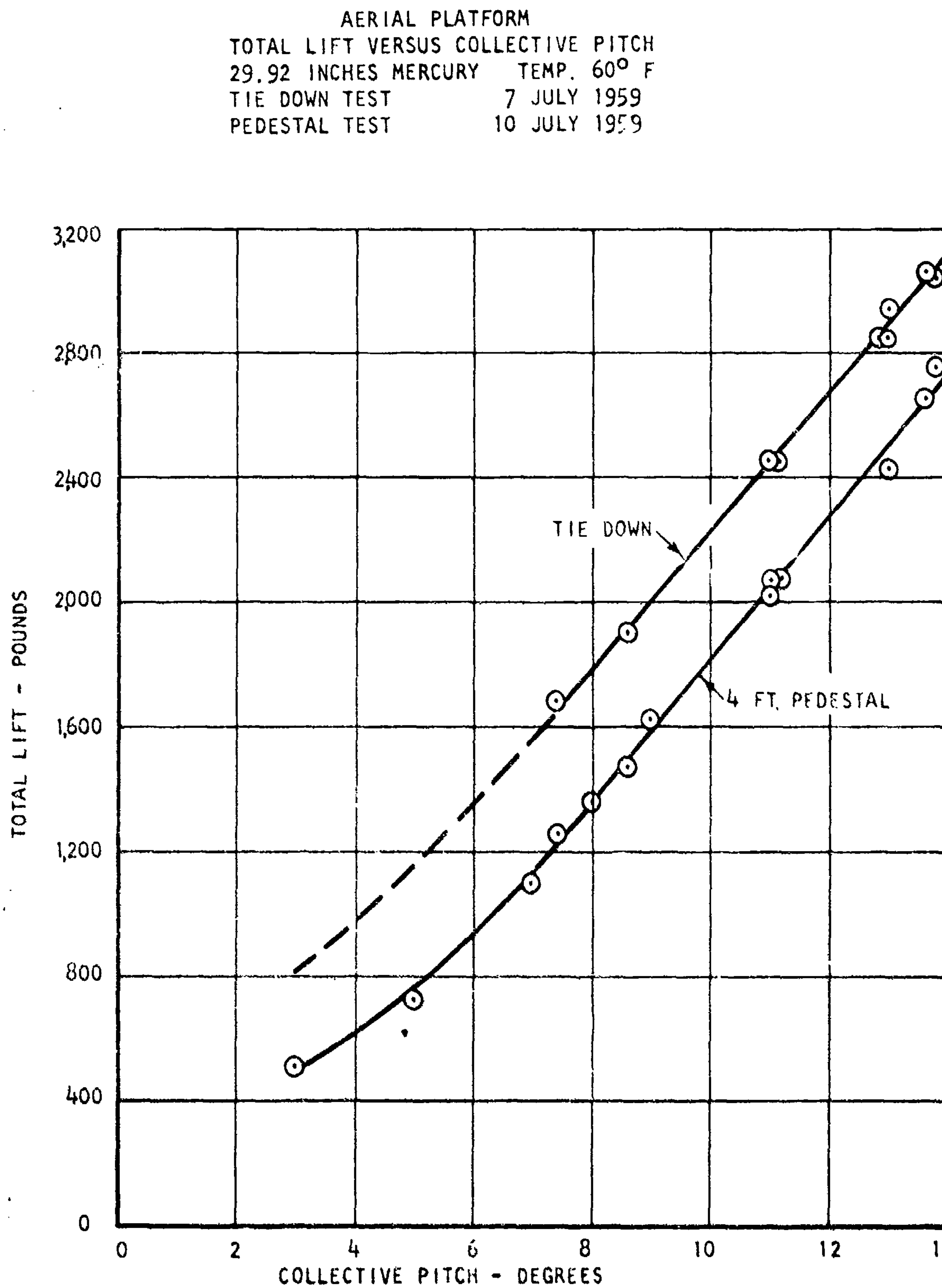

Figure 29 THRUST vs PROPELLER BLADE ANGLE

Tether Testing

Tether testing to determine the controllability of the Aerial Platform and to train the pilot for free flight was divided into 3 basic steps.

a. Nose wheel lift-off tests.
b. Main wheels lift-off tests.
c. Limited cable freedom tests.

Nose Wheel Lift-off Tests

The vehicle was tied securely to the test pad at the main landing gear. The tether cables to the forward propellers were rigged to permit as much as 10° pitch up of the entire vehicle. The pilot experimented with collective pitch and differential pitch to assess effectiveness of each in controlling pitch attitude. The pilot experienced no difficulty in controlling the pitch attitude in this single-degree-of-freedom set-up.

Main Wheels Lift-off

This test set-up was similar to that of the nose wheel lift-off tests, except in this case the main gear was free (within tether limits) while the nose wheel was secured to the test pad. In this test, both pitch and roll controls would be effective, for the set-up gave two degrees of freedom, i.e., pitch and roll. The pilot had no difficulty lifting the main gear clear of the ground. Assessment of pitch and roll control effectiveness was made difficult by the tendency of the vehicle to translate sidewards from even nominally small roll attitudes. This translation was sensed by the pilot as an unorthodox yaw-about-the-nose-wheel rather than a pure sidewards motion.

Minimum Freedom Tether Tests

On the initial trials the four cable parallelogram tether rig was adjusted to give only ten to twelve inches of cable freedom. Although the pilot was able to hold all three wheels off the ground for periods up to nine seconds, he found it very difficult to fly within the extremely small "sphere of freedom" allotted him.

These tests were re-run with successively greater cable freedom up to 18". The increased freedom materially assisted the pilot for it reduced the tendency to over-control in order to stabilize all maneuvers within the small sphere of freedom.

Maximum Freedom Tether Tests

Based on the above experience, tether cable freedom was extended to approximately 32 inches. The results of these tests were very encouraging, for although there were winds of 10 mph (gusts to 17 mph) the pilot was able to consistently improve his technique in flying the vehicle. Figure 30 shows these tether flights. In the last tests, the cable slack retracting bungee cord was softened. This reduced the influence of the cables on the vehicle. Again the pilot found a noticeable improvement in roll and pitch controllability. Individual flights (i.e., all three wheels continuously off the ground) ran to 60 or more seconds.

A telemetering data link was used for all tethered flights. Vehicle attitude in pitch, roll and yaw, attitude change rates, control stick positions, blade angle positions and accelerometers were all recorded through this radio link. Post-flight study of vehicle attitude and control input traces indicated that the vehicle responded very closely to command inputs 16mm motion pictures were taken of all tether runs. Study of these films and the telemetry records substantiated the pilot's observations about his being in control of the vehicle. It is significant to note that this entire tether flight program was conducted without assistance from any stability augmenting equipment. The stability augmentor had been installed on the vehicle, but deliberately left inoperative, to assess vehicle behavior without the augmentor, representing flight conditions if the augmentor failed. Success of these tests without the augmentor led to completion of the program without ever having used the augmentor.

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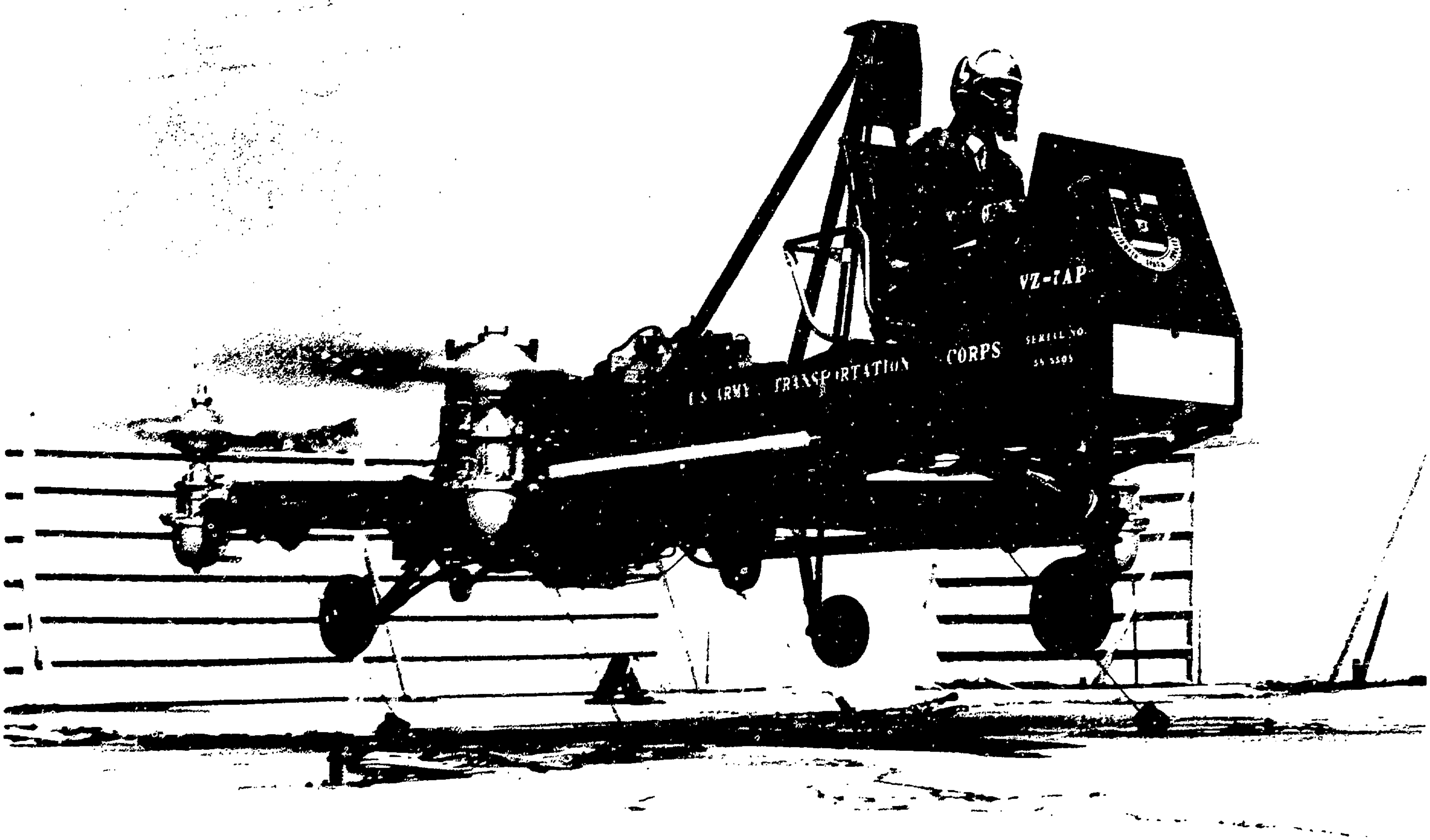


5. FREE FLIGHT TESTING

General Summary

The flight test program on the VZ-7AP involved 102 separate flights starting on 3 September 1959 and ending on 26 January 1960. These flights progressively established the excellent flight characteristics of this vehicle.

All of the test flying was done by Mr. C. Roger Gardner, Engineering Test Pilot for Wright Aeronautical Division of the Curtiss-Wright Corporation at Edwards Air Force Base, California. It is significant to note that Mr. Gardner has very limited helicopter flight experience (although well experienced in fixed wing aircraft). His total helicopter time prior to the VZ-7AP flight program was less than 40 hours.

It is also significant to note that all flying of the VZ-7AP was done without assistance from any stability augmenting equipment.

The free flight test program was arranged to progressively explore and determine the flight characteristics of the Aerial Platform. Initial flights were quite modest in scope of maneuvers while later flights involved very rigorous maneuvering. Each flight for official data record purposes was preceded by one or more practice flights to take advantage of the continually increasing pilot skill and confidence levels.

Flight test data was recorded by telemetry techniques and documented by photo means to assure accurate and reliable flight test data.

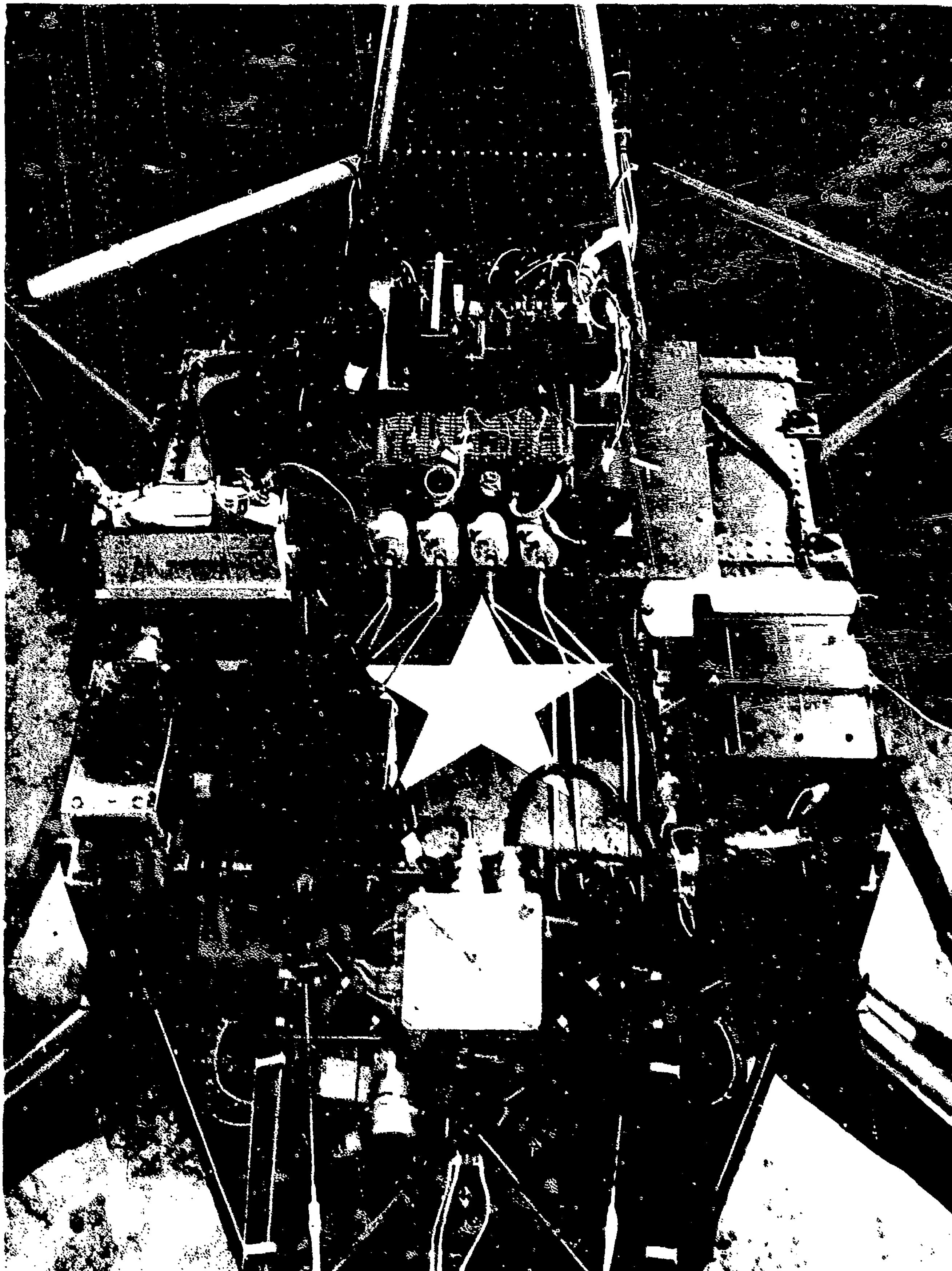
Instrumentation

A telemetry system was used throughout Phase III testing as the primary data link. The system had been installed and checked out in Phase II (Reference 32) where it was used for data purposes in the later ground tests.

The on-board instrumentation package consists of a PWM/FM/FM system with three continuously operating subcarrier channels and one subcarrier channel which was commutated with a 43 channel sampling switch. Each channel was sampled 20 times per second. The transmitter of this instrumentation package is the Tele-Dynamics Model 840 B1 operating on the 219 megacycle range with an average output of 4.5 watts. The subcarrier oscillators are EMR 75 A voltage controlled operating on the IRWG channel of 70 KC for the commutated channels and 14.5 KC for the continuous channels. The transmitter units have been assembled into a 24 channel Utility Telemetry Package. This package, along with some of the transducers, is shown in Figure 31, as installed on the Aerial Platform. A block diagram of the system is shown in Figure 32.

ASCOP, Type M, ground station equipment was used to receive and record the incoming signals. The test data was taped utilizing an Ampex recorder and simultaneously recorded with direct writing Sanborn systems. Conversation between pilot and ground crew was also recorded with the Ampex system. The Sanborn recorder, as built into the ASCOP system, has a maximum calibration error of 0.5 mm over a total recordable range of 50 mm. Calibration checks were made at appropriate intervals to assure accuracy of recorded data.

A list of the on-board equipment including values of expected accuracy of the finished Sanborn record is presented in Table 4.



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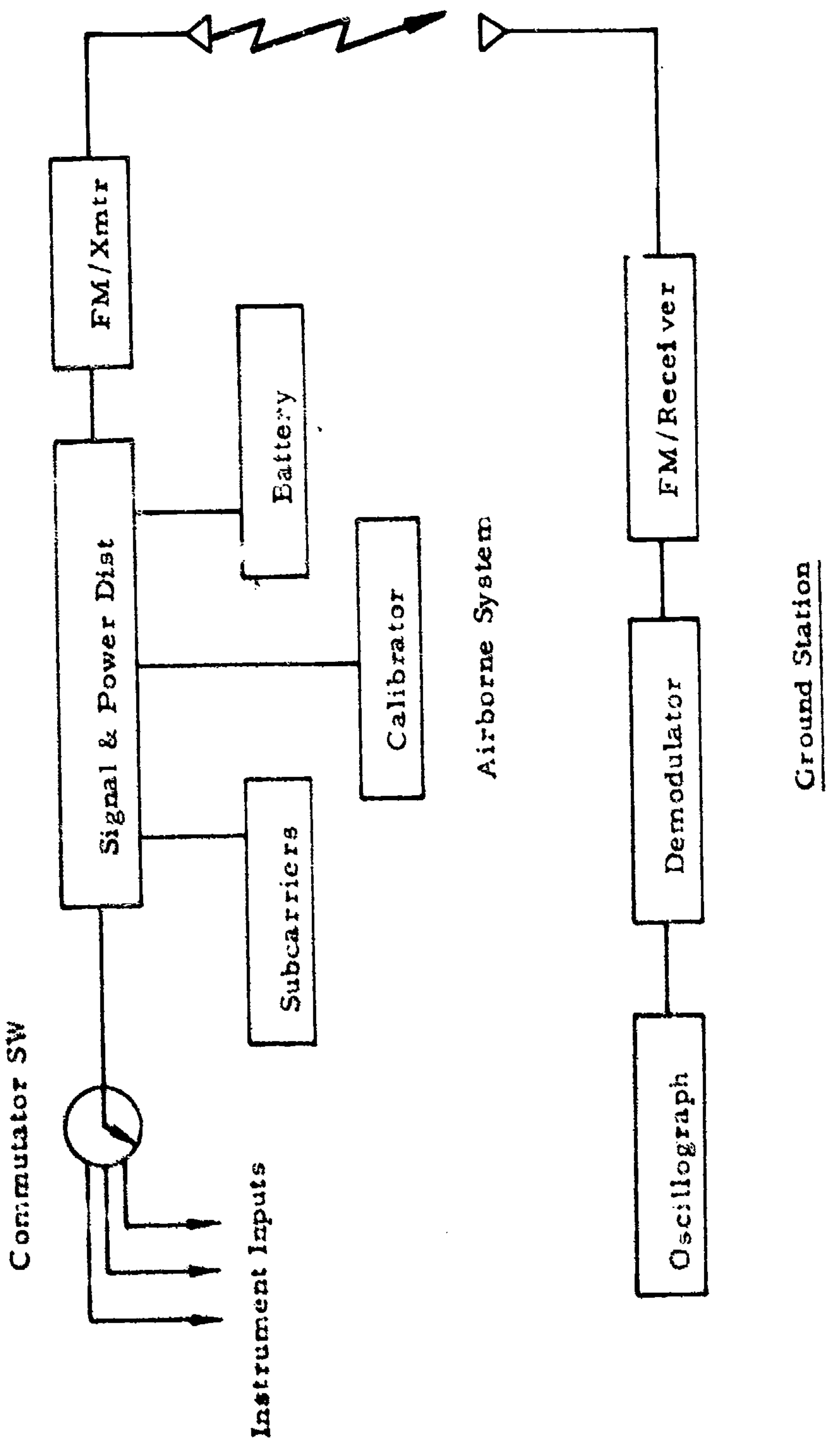

Figure 32. BLOCK DIAGRAM, TELEMETRY SYSTEM

TABLE 4

ON-BOARD INSTRUMENTATION EQUIPMENT

Instrument	Measurement	Range	Accuracy*
Accelerometer	Vertical accel.	±5 G's	±.1 G's
Accelerometer	Lateral accel.	±2 G's	±.04 G's
Accelerometer	Long. accel.	±2G's	±.04 G's
Accelerometer	Vibration amplitude	±40 G's	±.8 G's
Attitude Ref Gyro	Roll position	±90°	±2°
Attitude Ref Gyro	Pitch position	±90°	±2°
Attitude Ref Gyro	Yaw position	±90°	±2°
Attitude Ref Gyro	Yaw position	±90°	±2°
Potentiometer	Pitch stick position	±15°	±0.3°
Potentiometer	Roll stick position	±15°	±0.3°
Potentiometer	Coll. Stick position	50 to 100°	±0.1°
Potentiometer	Prop. pitch RF	7 to 17°	±0.2°
Potentiometer	Prop. pitch RR	3 to 13°	±0.2°
Potentiometer	Prop. pitch LR	3 to 13°	±0.2°
Potentiometer	Prop. pitch LF	7 to 17°	±0.2°
Potentiometer	Rudder position	±20°	±0.6°
Press. Trans.	In-take Velocity UR	±0.5 psi	±.01 psi
Press. Trans.	In-take Velocity LR	±0.5 psi	±.01 psi
Press. Trans.	In-take Velocity UL	±0.5 psi	±.01 psi
Press. Trans.	In-take Velocity LL	±0.5 psi	±.01 psi
Rate Gyro	Pitch rate	±90°/sec	±2°/sec
Rate Gyro	Roll Rate	±90°/sec	±2°/sec
Rate Gyro	Yaw rate	±90°/sec	±2°/sec

*Accuracy values are those expected through the entire telemetry data system, including presentation on paper tapes.

First Free Flight Tests

The first free flight tests were successfully conducted on 3 September 1959 at the Santa Barbara Municipal Airport. See Figure 33. Two groups of flights totalling 29 minutes engine time were made. Individual flights lasted up to four minutes. The pilot demonstrated closely controlled hovering flight and experimented very briefly with forward and rearward translations. The pilot reported that the free flight techniques were much easier than those required for "tether flying".

In subsequent early flights the pilot experimented much more freely with translational flights, both forward, backwards and sidewards with good control of attitude and speed. Pilot skill in handling the vehicle improved with each flight. Pilot fatigue did not appear to be a problem. Normal flight altitude was approximately 3 to 4 feet. Altitudes of 8 to 10 feet were achieved on a momentary basis. The vehicle exhibited good response to roll and pitch commands. Yaw control was found to be only slightly effective in calm air and totally ineffective in winds of 7 mph.

The substandard yaw control effectiveness was eliminated by the addition of yaw vanes mounted beneath the rear propellers as shown in Figure 15 and 34. With these vanes, the pilot was able to maneuver and hold the vehicle on any heading in winds of 16 mph (gusts to 20 mph). See Yaw Stability and Control Modifications, page 121.

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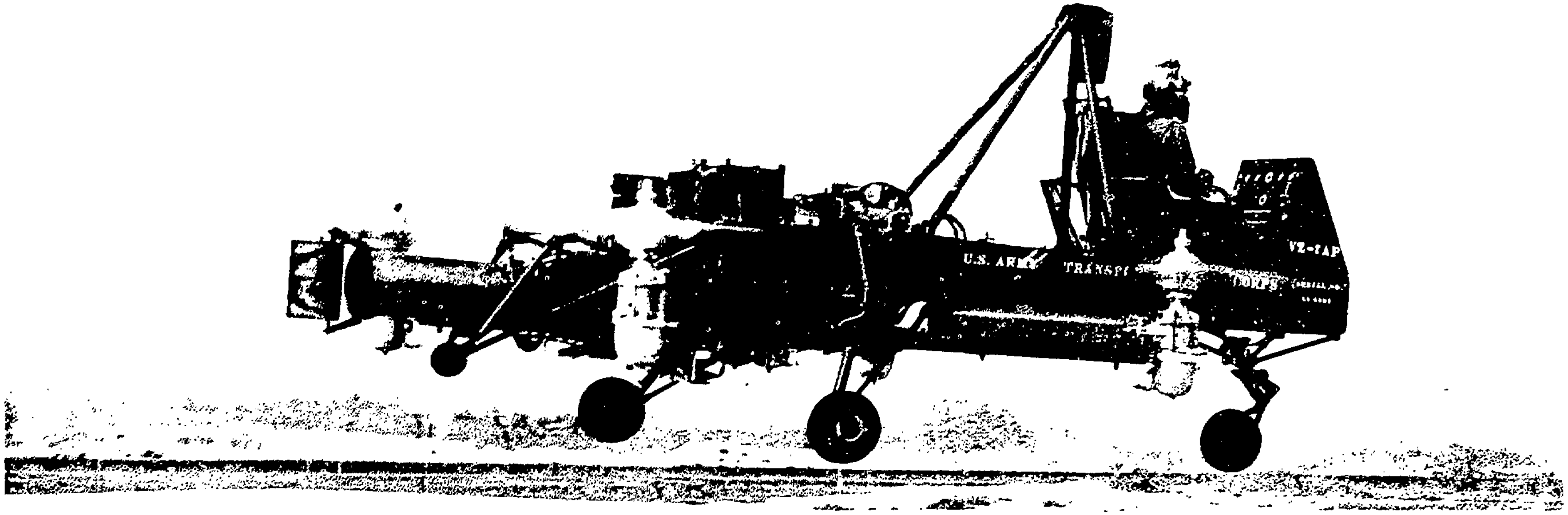

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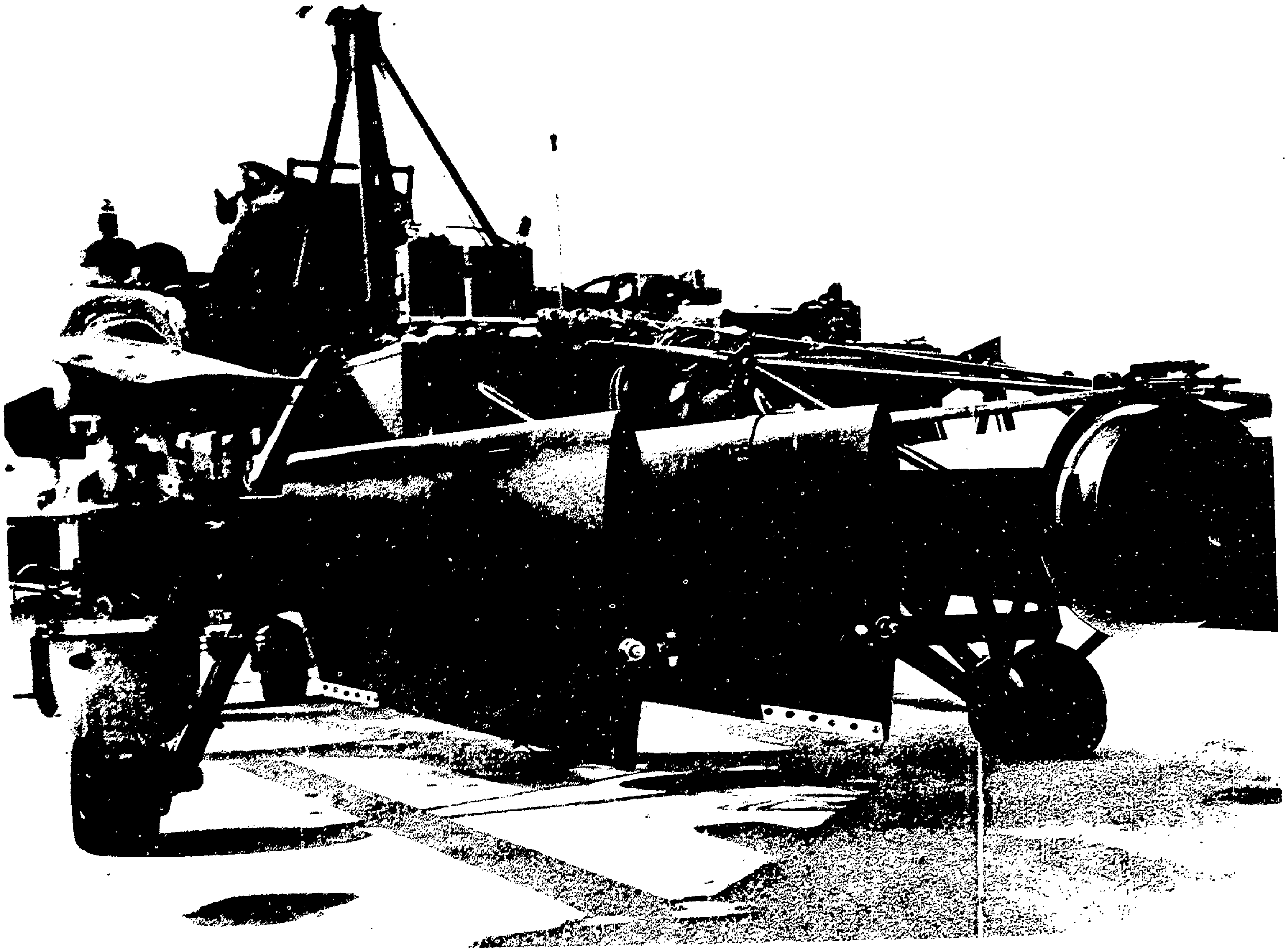


Yaw Response Tests

The effectiveness of the yaw control was tested by having the pilot displace the rudder pedals while hovering, both from the rest condition and while in an initial turning condition. Figure 35 shows a typical time history of a full pedal deflection maneuver and recovery starting from rest. The yaw angular acceleration data obtained from 15 such maneuvers is presented in Figure 36, as a function of effective rudder deflection $\left(\delta_R \pm K \dot{\psi} \frac{\text{Deg.}}{\text{Sec.}}\right)$

where δ_R = actual rudder deflection

$\dot{\psi}$ = yaw rate, deg/sec

K = a theoretcially derived constant used to correct for variation in rudder effectiveness due to vehicle angular velocity.

Correlation of the yaw angular acceleration data with rudder position showed that the yaw angular velocity damping was negligible, but that rudder effectiveness was dependent upon the yaw vehicle angular velocity. This effect can be seen in Figure 37 which depicts the effective angle of attack of the rudder at both zero and finite angular velocity, and realizing that the rudders are at some distance r from the center of rotation of the vehicle in yaw.

Thus it is quite apparent that the rudder is less effective when deflected in a direction to give a moment in the same direction as the turning rate, and more effective when deflected to give a moment opposite to the direction of the turning rate. Consequently, all yaw control effectiveness data have been correlated against the parameter $\delta_R - K\dot{\psi}$, where K is the effective length between the rudder center of pressure and the turning center divided by the effective rotor jet velocity. The value of K was determined from the experimental data with the rudder both against and with the turn and was found to have a magnitude of 0.375. It can be seen from Figure 36 that this technique results in nearly linear yaw angular acceleration with effective rudder deflection. At no time during the flight test program (after the addition of yaw vanes) did the pilot experience any difficulty in controlling the yaw attitude.

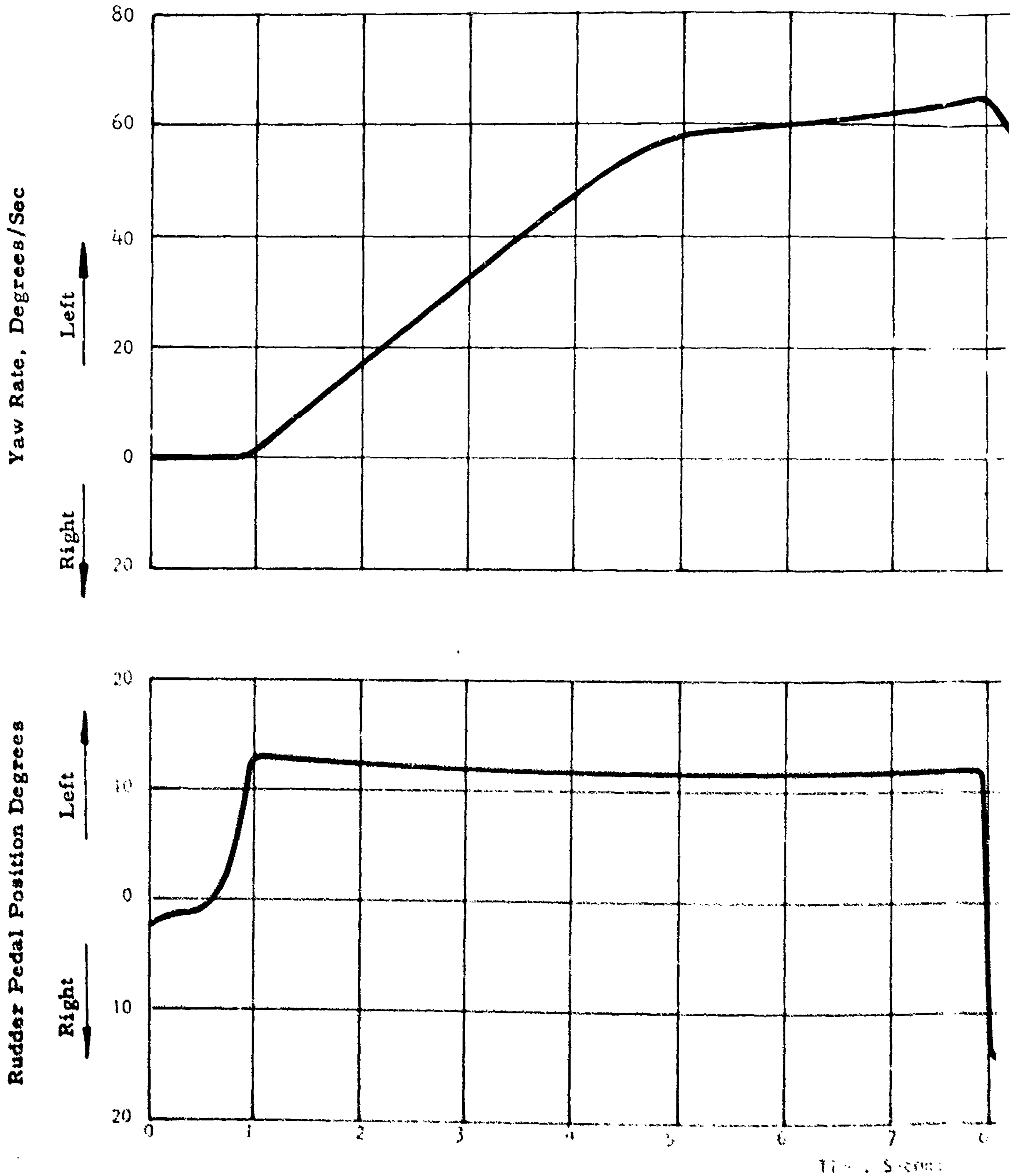
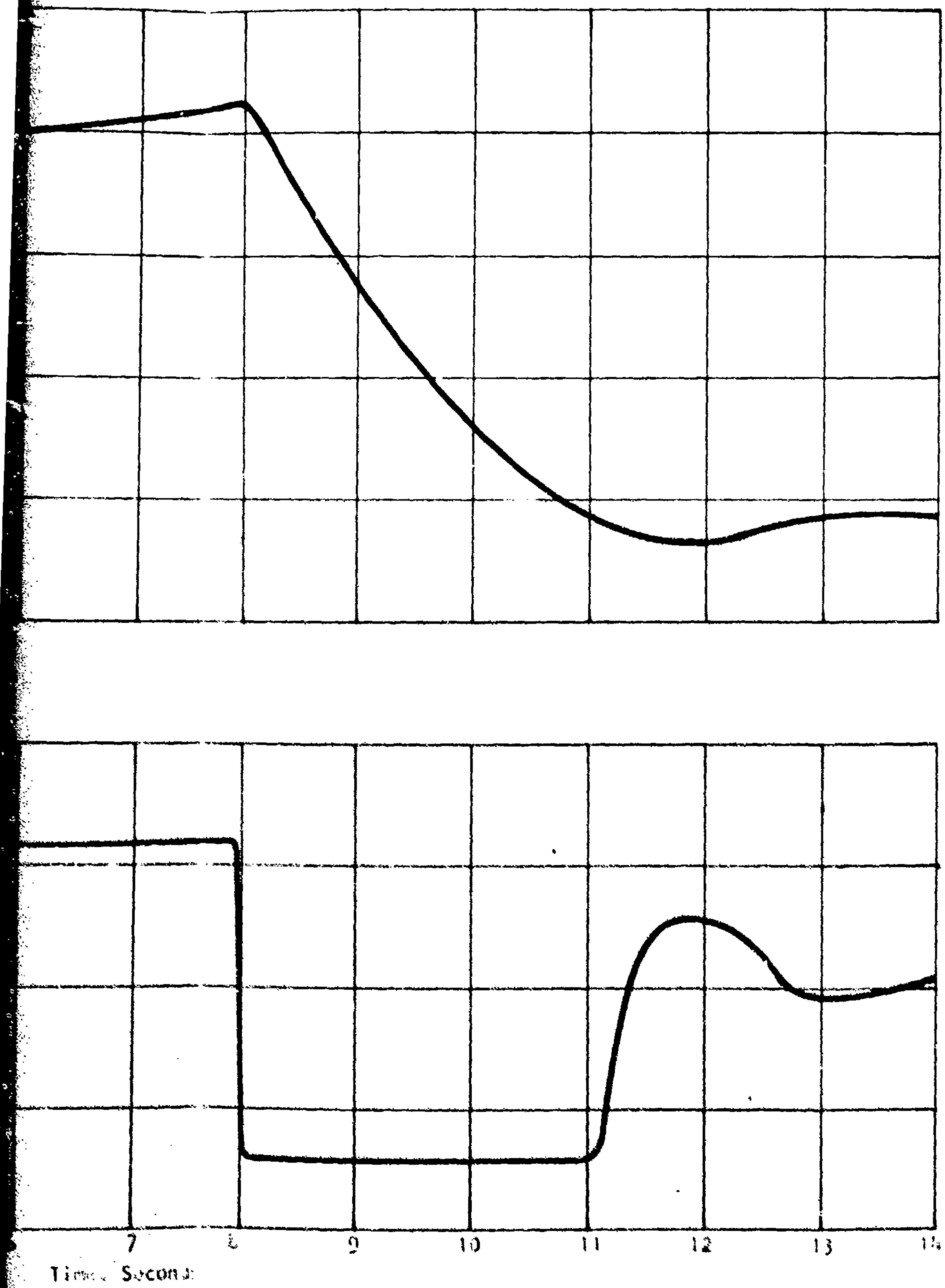


Figure 35 TIM HISTORY, TYPICAL YAW MANEUVER,

Ambient Conditions
Sea Level
62° F



YAW MANEUVER, HOVERING

Ambient Conditions:
Sea Level
68°F
6 ft Average Altitude
Fixed Fin not in Place

● Denotes accelerations from rest position
○ Denotes accelerations from finite angular velocities

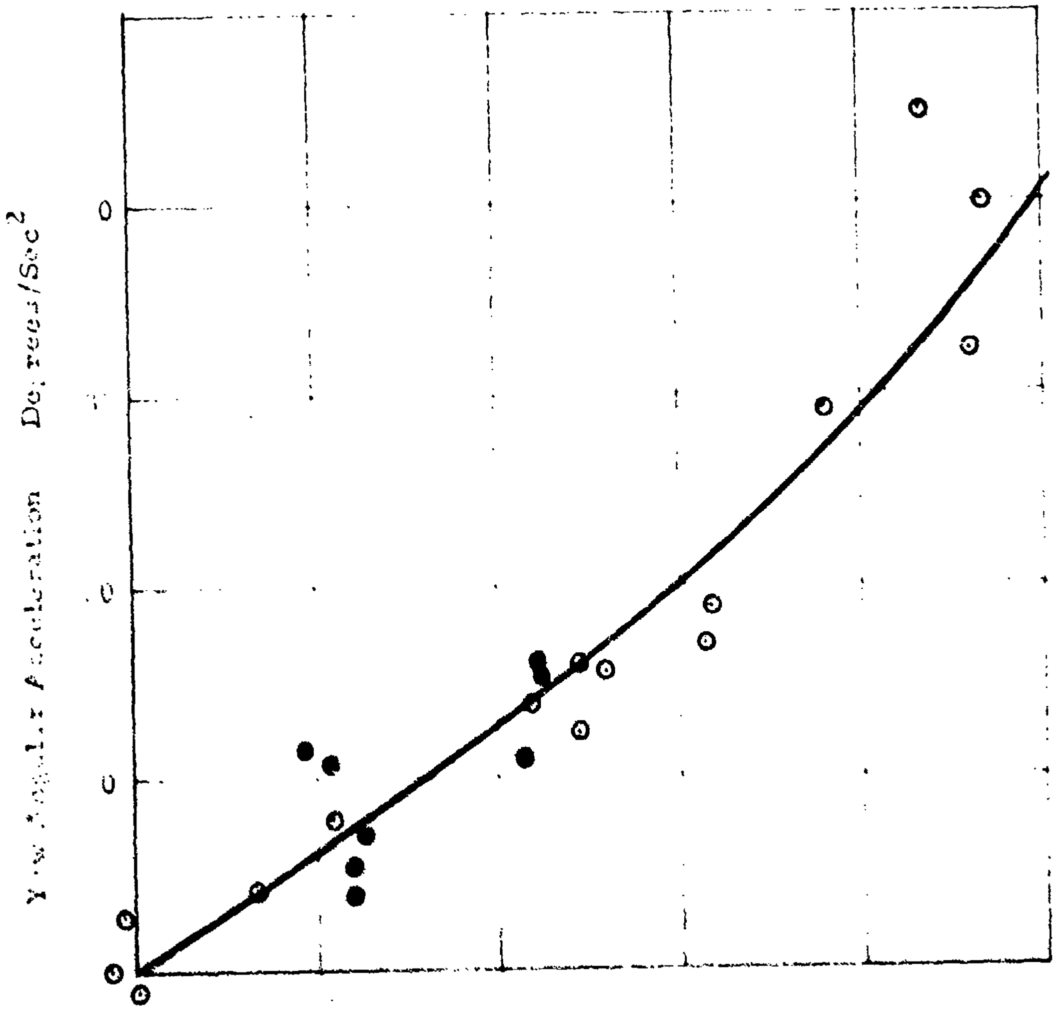


Figure 56 RUDDER CONTROL EFFECTIVENESS, HOVERING

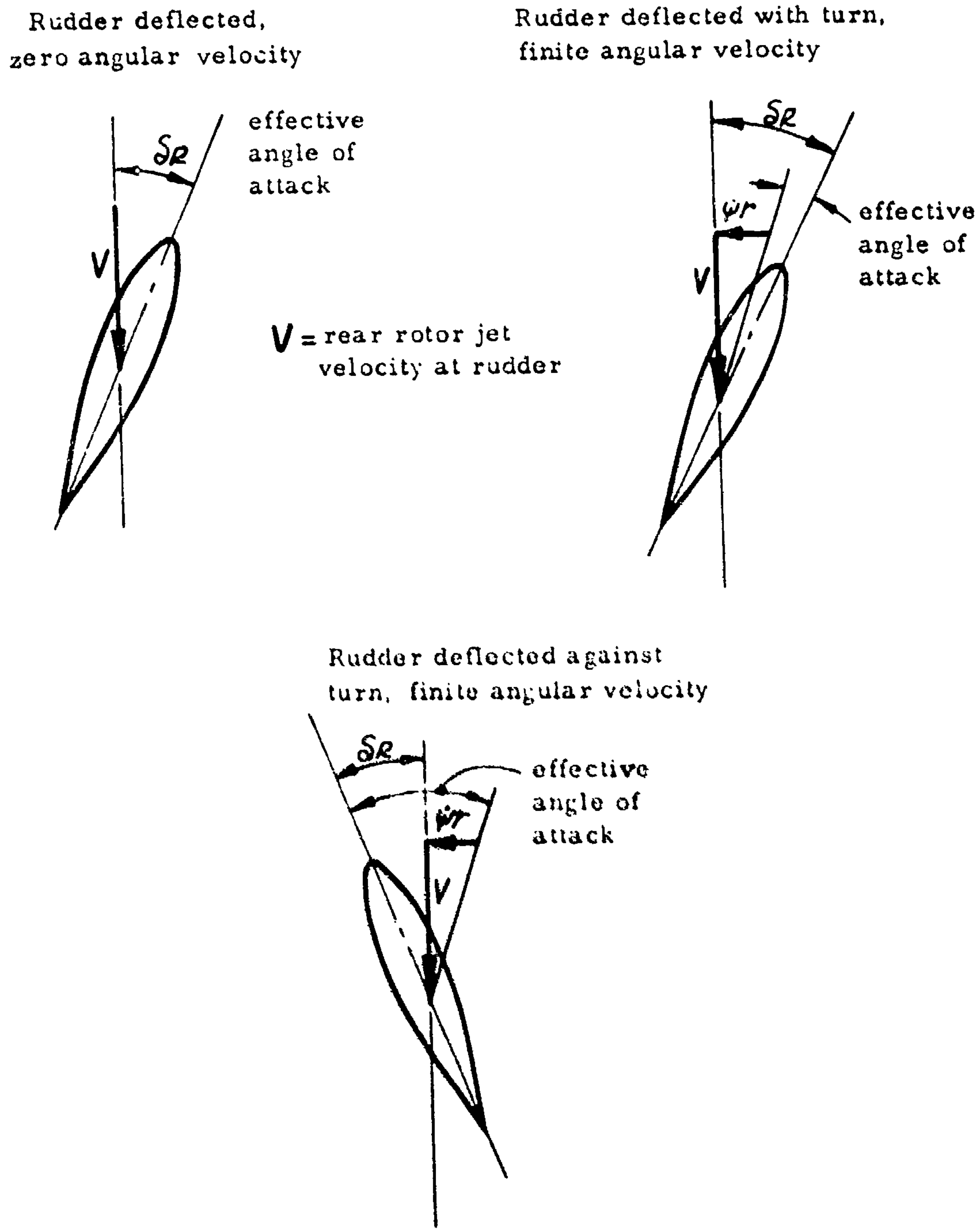


Figure 37 RUDDER EFFECTIVENESS DIAGRAMS

Directional Stability Tests

The directional stability of the Aerial Platform with *vertical fin was determined by flying the vehicle at a nearly constant angle of yaw, measured from the runway centerline, while moving along the runway. Speed was determined by means of a pace car. Yaw attitude (measured by a directional gyro) and rudder position were recorded by the use of the telemetry system. These data were correlated by plotting yaw angle per degree rudder pedal deflection versus forward speed, Figure 38. It can be seen that less yaw is developed per degree pedal movement as the forward velocity increases. This is a result of the directional stability of the vehicle, which provides a stabilizing moment counteracting the yaw control moments. Since the stabilizing moment increases approximately as the square of the forward velocity, the maximum yaw decreases nearly inversely to the square of the velocity. This tendency is clearly shown in Figure 38.

Considering the results of these yaw runs, using a fin reference area of 12.5 square feet and a reference length of eight feet (the distance from the fin center of pressure to the vehicle center of gravity) a value for the fin side force coefficient as a function of the yawing angle (C_{yB}) was calculated to be .08 per degree. This value is higher than the estimated value of 0.05 used in the fin design calculations and indicates that less conservative assumptions could be used in establishing fin area criteria for future configurations.

*See Yaw Stability & Control Modifications, page 121

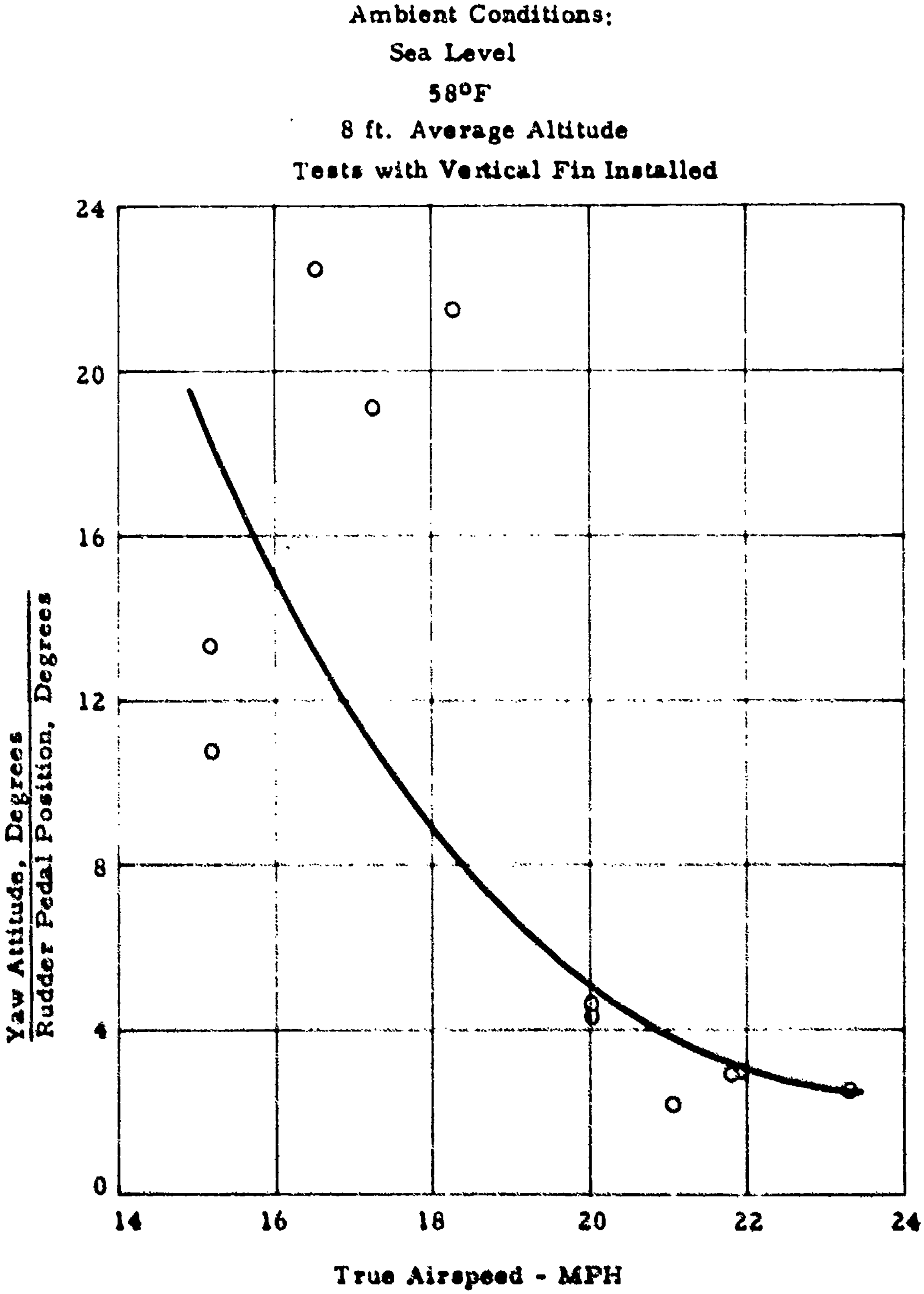


Figure 38 DIRECTIONAL STABILITY AT FORWARD SPEEDS

Roll Response Tests

Roll response and control effectiveness were determined in a series of tests wherein the pilot, starting altitude from a hovering position at approximately 10 feet, produced a vigorous roll maneuver by a rapid lateral displacement of the differential control stick in the command direction followed by equally vigorous recovery command motions.

Roll attitude and roll rate intelligence was obtained from position and rate gyros, respectively, and telemetered to the ground station. Stick position was obtained from a potentiometer geared to the main control stick torque tube and recorded through the same telemetry system.

A typical time history of one of these maneuvers is shown in Figure 39. Roll angular acceleration as a function of stick deflection for three of these runs is presented in Figure 40. Superimposed on this plot are values for maximum allowable accelerations and desired accelerations per inch of stick travel, for usual flight conditions, as described in an unreleased report by N.A.S.A., Langley and presented at the 28th Annual I.A.S. Meeting January 25-27, 1960, by Robert J. Tapscott in a paper entitled "Criteria for Control and Response Characteristics of Helicopters and V.T.O.L. Aircraft in Hovering and Low Speed Flight". It can be seen that the roll control effectiveness is quite close to the acceleration versus control input slope considered desirable by the NASA. The NASA requirement is stated in terms of roll deflection per inch of stick travel at the end of 0.5 seconds. This equivalent acceleration per stick deflection has been assumed to also be valid for greater control deflections for evaluation of the test data.

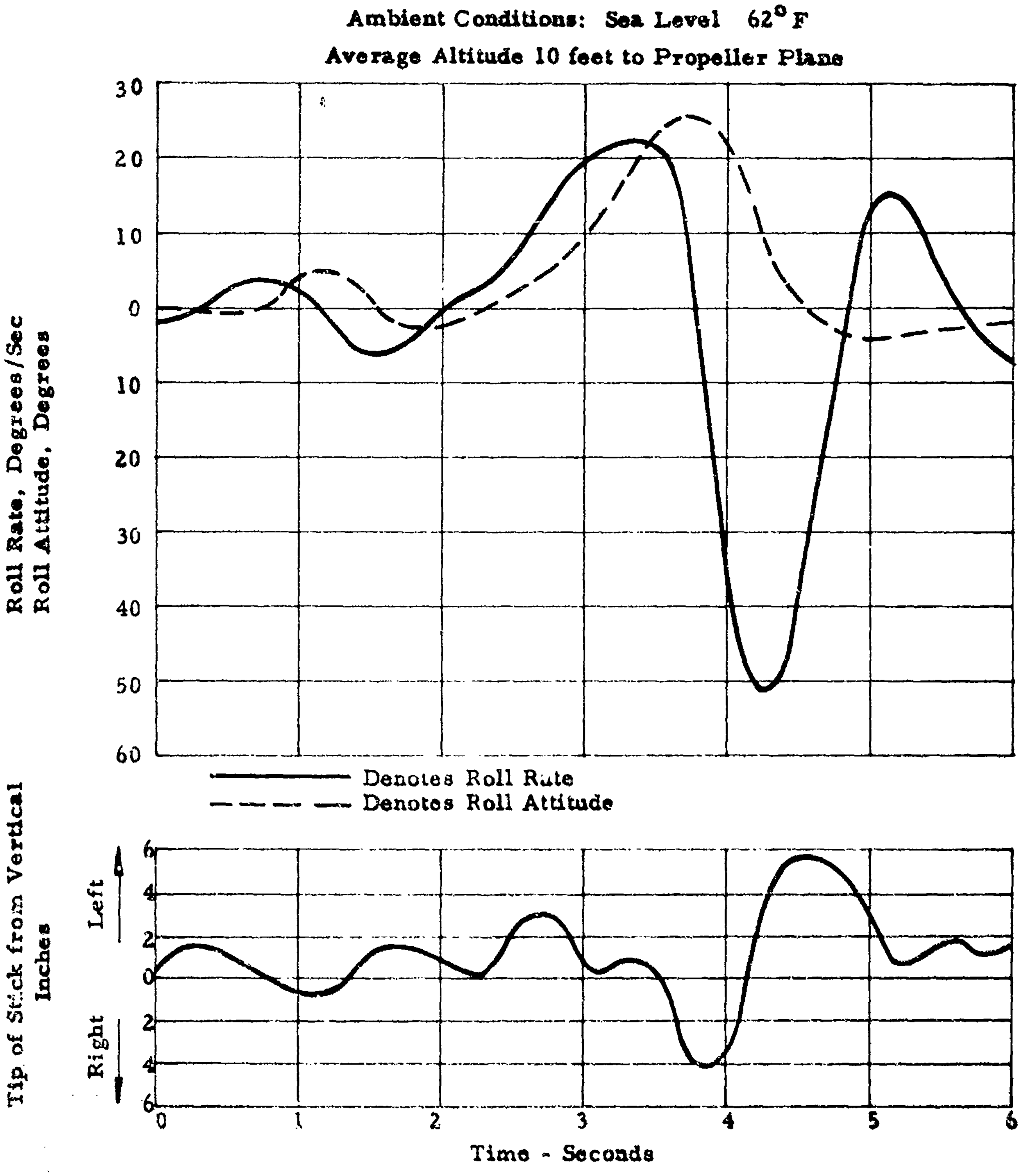


Figure 39 TIME HISTORY, TYPICAL ROLL MANEUVER, HOVERING

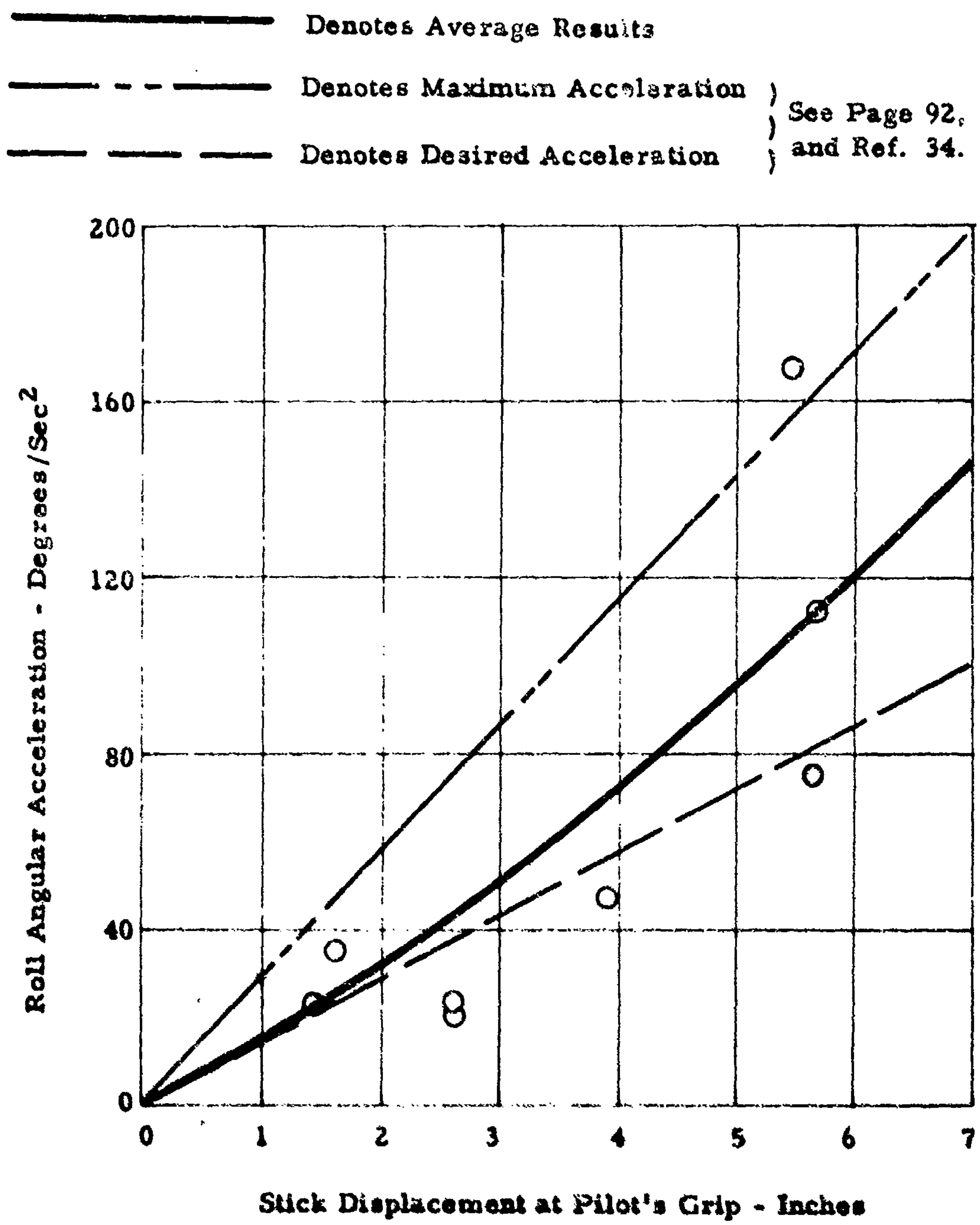


Figure 40 ROLL CONTROL EFFECTIVENESS

Pitch Response Tests

Pitch response and control effectiveness were determined in a series of tests wherein the pilot, starting from a hovering position at approximately 10 feet altitude, produced a vigorous nose down pitching maneuver by rapidly displacing the differential control stick in the forward direction, quickly followed by equally vigorous stick motions to affect recovery of attitude.

Vehicle attitude, rate and stick positions were telemetered to the ground station.

A typical time history of one of these maneuvers is shown in Figure 41 . Pitch angular acceleration as a function of pitch stick deflection for several test runs is recorded in Figure 42 . Here again, maximum and desired values for control effectiveness, as determined from the forementioned sources, are included for comparison purposes. The pitch control effectiveness of the Aerial Platform is somewhat greater than that considered desirable by the NASA criteria, however, the pitch control must have sufficient effectiveness to permit the pilot to trim the vehicle at various forward velocities and with varying center of gravity positions. The addition of the trim requirements to the pitch control necessitates that the pitch control have this increased sensitivity or possibly a separate trim device. The magnitude of the trim change required with changes in velocity is presented with other data from speed runs. See Figure 45.

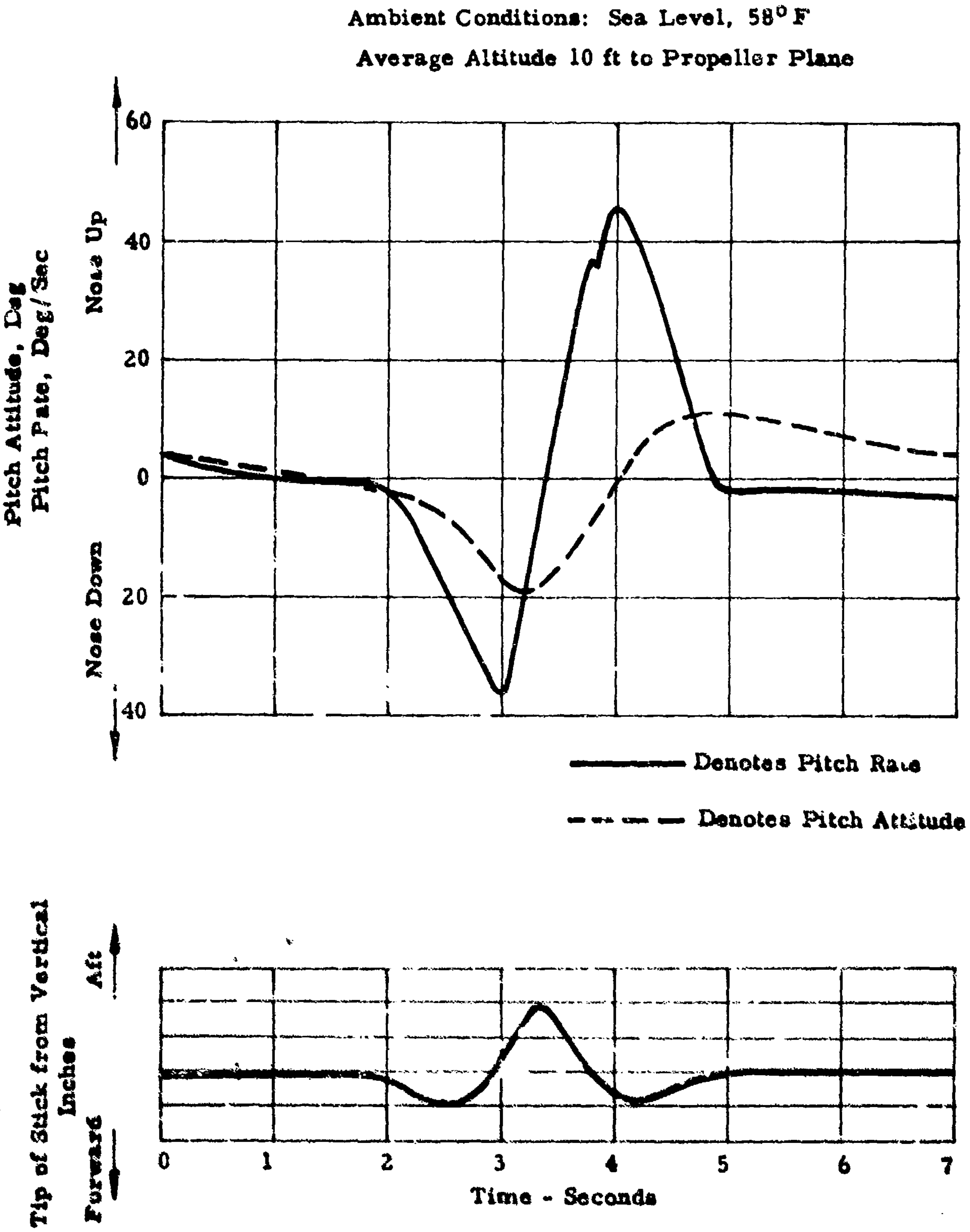


Figure 41 TIME HISTORY, TYPICAL PITCH MANEUVER, HOVERING

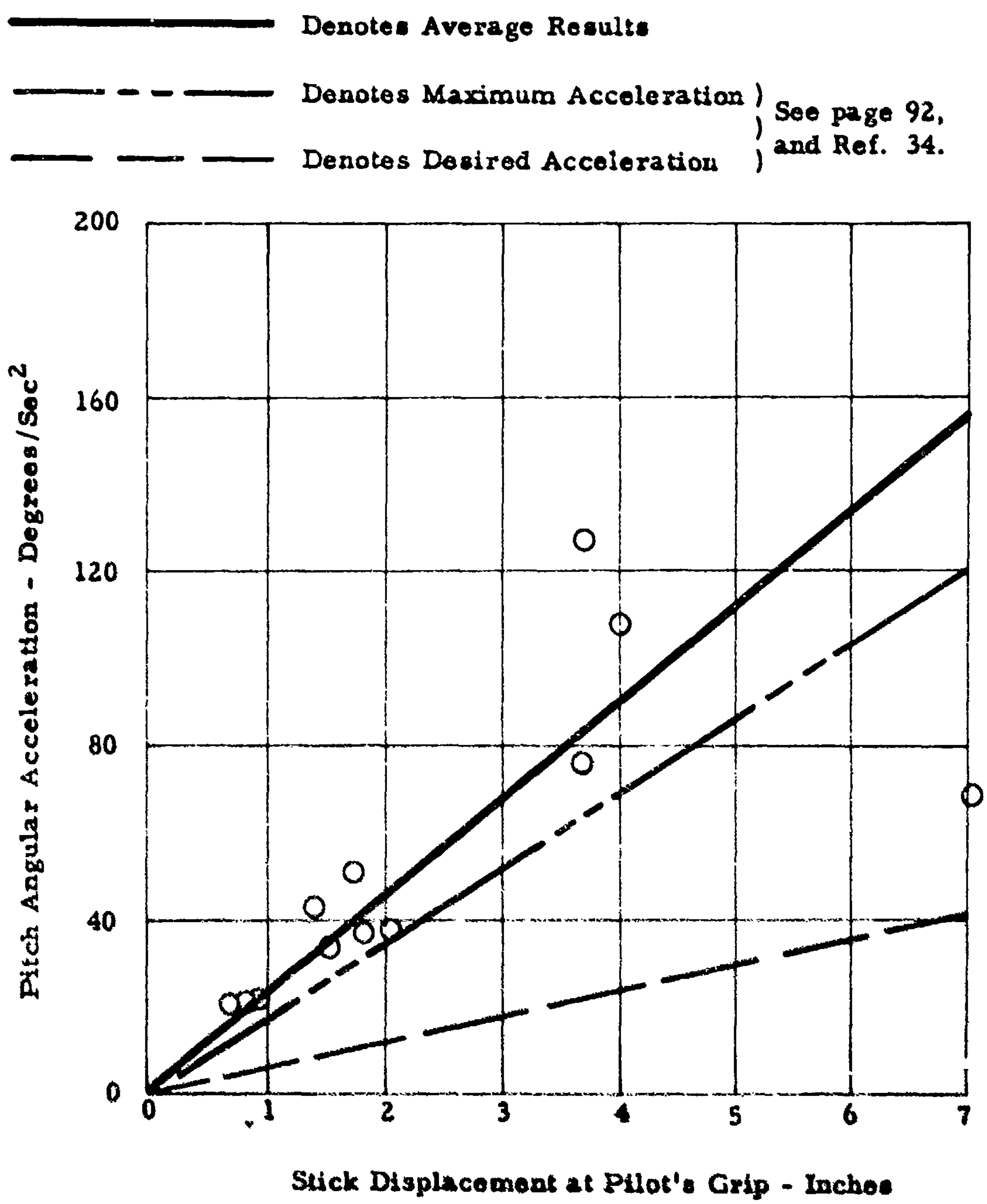


Figure 42 PITCH CONTROL EFFECTIVENESS

Pitch Dynamic Stability Tests

To determine the vehicle pitch dynamic stability without corrective action inputs from the pilot, a series of fixed stick stability runs were made. The vehicle was equipped with a test rig by which the pilot could lock the differential control stick against rearward motion while retaining stick freedom in the lateral and forward directions. After setting up steady hovering conditions at approximately 10-12 feet altitude the pilot could set the stick lock device in the hovering trim position and then hold the stick firmly against the stop device. With this set-up the pilot could provide different command inputs, such as a sharp nose down pulse, and then ride out the attendant oscillations with the stick locked in the original hovering position.

Since the maximum test altitude was limited for safety reasons, the pilot was obliged to take corrective action (by unlocking or overpowering the lock device) before the vehicle could complete more than one cycle with the stick fixed.

Excellent records of vehicle behavior were obtained through the telemetering system. Vehicle attitude and rate intelligence was sensed by position and rate gyros. Pilot action, i.e., stick displacement, was monitored by a potentiometer mounted on the main control torque tube.

Time histories of typical fixed-stick pitch maneuvers are shown in Figures 43 and 44 . Figure 43 shows a time history of a fixed stick run following a sharp nose down pulse (1 inch stick displacement; total time .5 seconds). Figure 44 shows a time history of a fixed stick run following a nose up-nose down sine wave input (plus and minus .3 inch stick displacement; total time, 3.5 seconds).

Figure 44 shows a period of eight seconds and a divergence rate of approximately 2.6 per cycle as compared to a predicted period of 24 seconds and a predicted divergence rate of 1.2 per cycle for unshrouded propellers. Using these fixed-stick flight test data, new values for M_u and M_q (coefficient of pitch angular acceleration due to forward velocity and coefficient of pitch angular acceleration due to pitch rate, respectively) have been determined. A comparison of these values with those predicted prior to the flight tests is shown in Table 5 . It appears that the predicted M_q value was realistic but that the M_{μ} value is much greater than anticipated. This may be due, in part, to a carryover effect of the propellers on the vehicle structure but more probably due to greater interference between the front and rear propellers than what could be estimated from available data.

TABLE 5

A Comparison of Predicted and Experimentally Determined Pitch Stability Derivatives

Derivatives	Predicted Hovering Shrouded Propellers	Predicted Hovering Unshrouded Propellers	Flight Test
$X_u \frac{ft/sec^2}{ft/sec}$	-.15	-.0365	-.0365*
$X_\theta \frac{ft/sec^2}{deg.}$	-.56	-.563	-.563*
$M_u \frac{deg/sec^2}{ft/sec}$	+3.70	+.206	+2.61
$M_{\dot\theta} \frac{deg/sec^2}{deg/sec}$	-.195	-1.59	-2.04

*Assumed Values

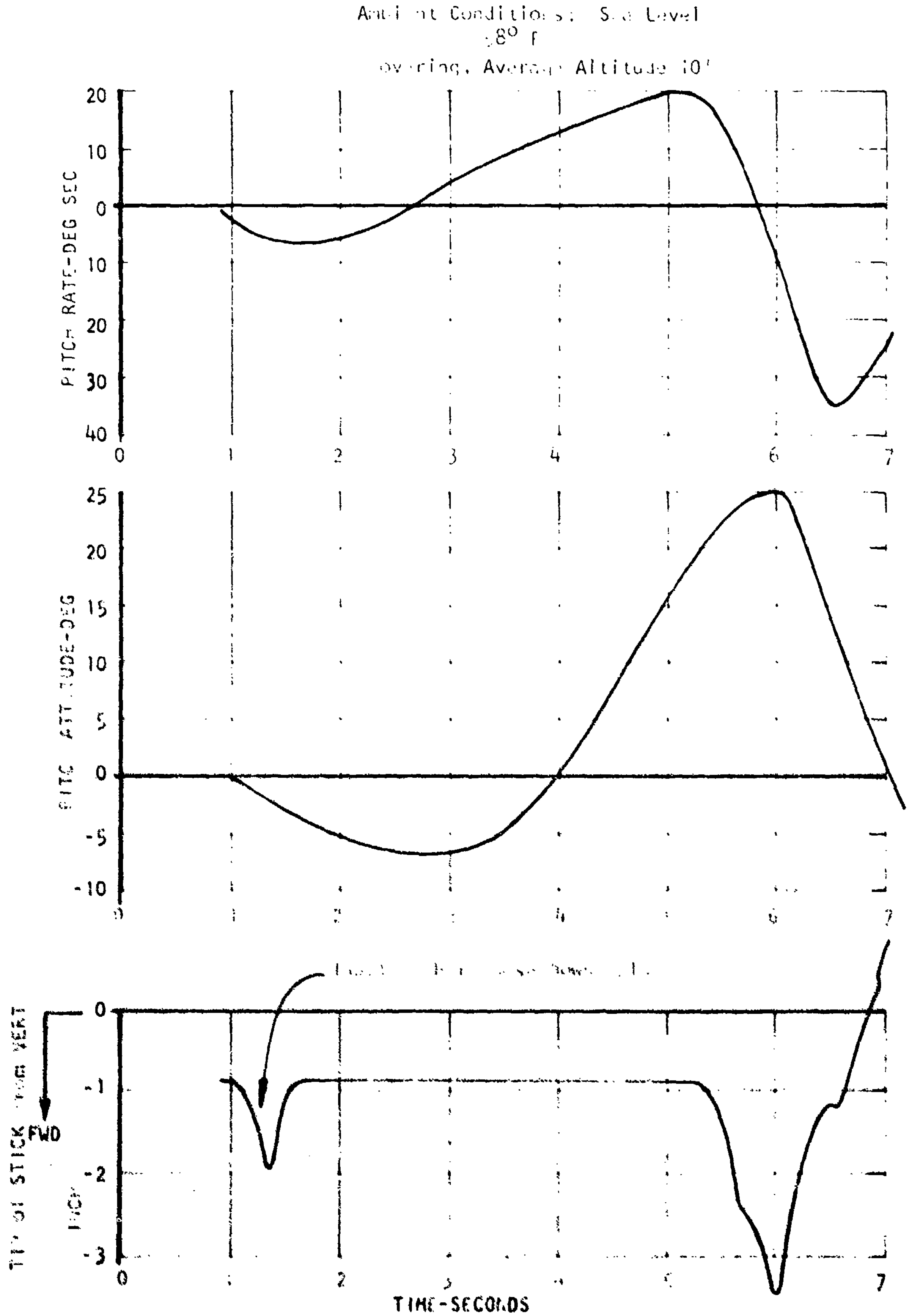


Figure 43: TIME HISTORY, FIXED STICK STABILITY

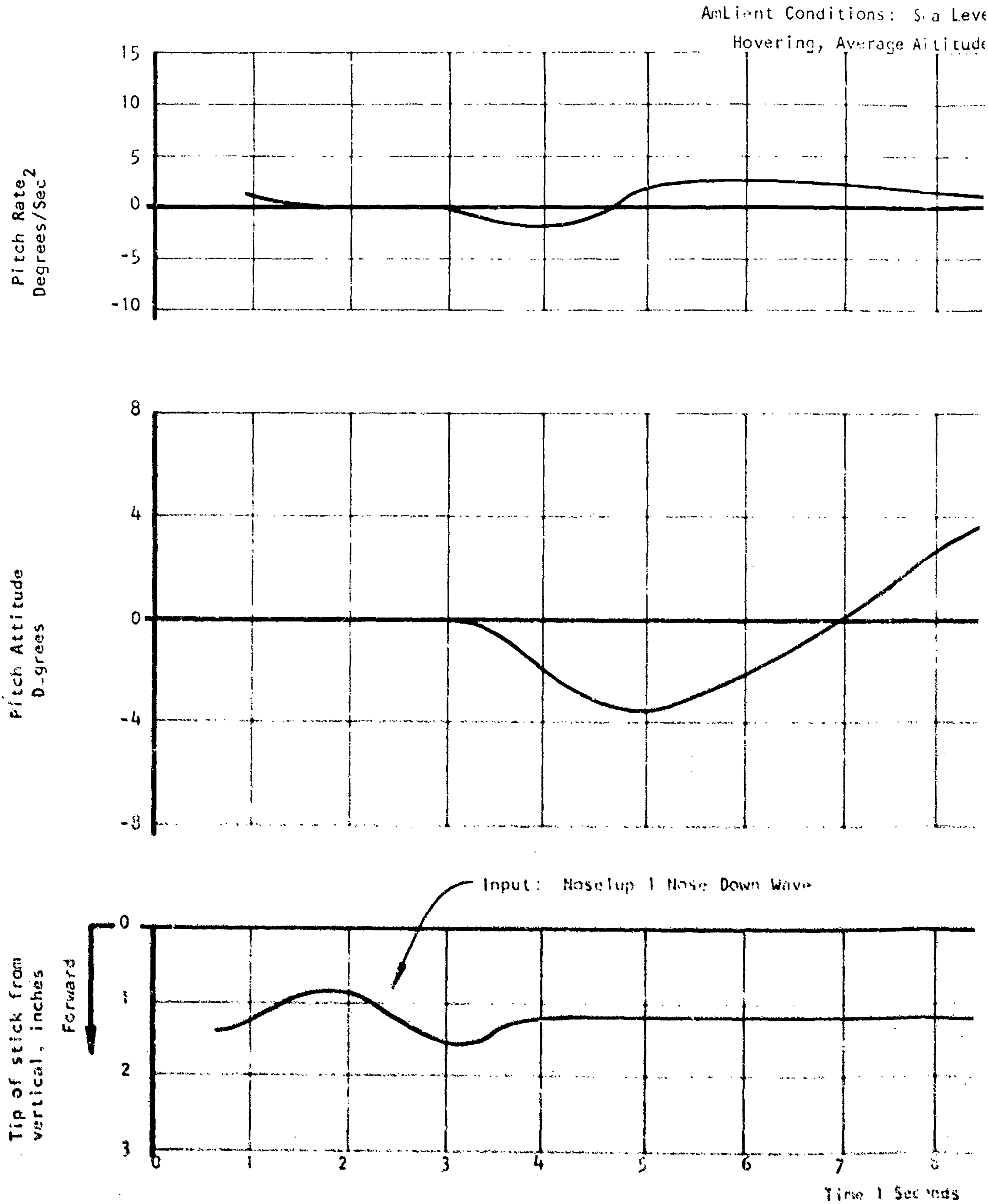
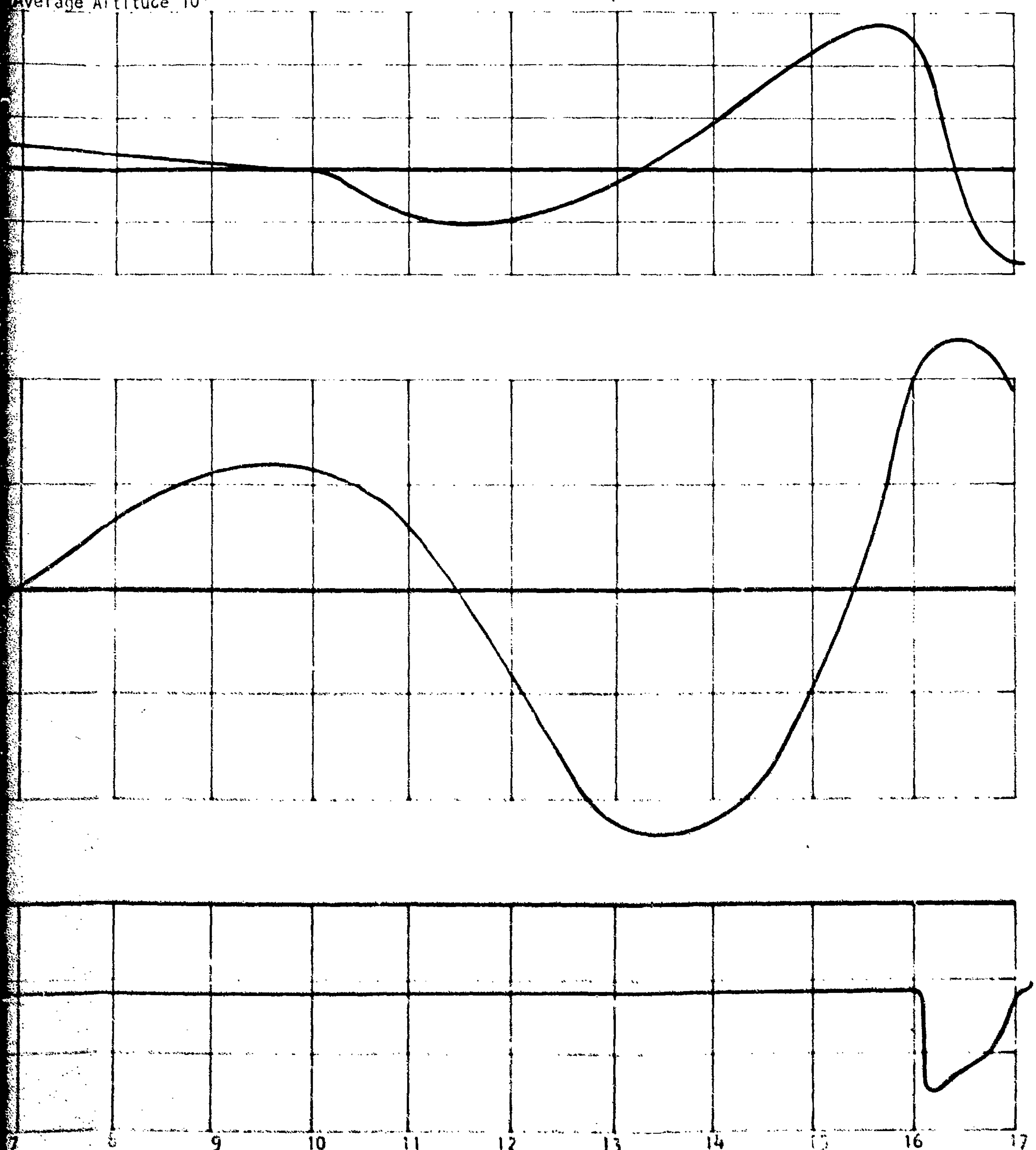


Figure 44 TIME HISTORY, FIXED STICK STABILITY

tions: Sea Level, 58° F

Average Altitude 10'



ime 1 Sec nds

Speed Tests

Tests at various forward speeds were made to determine vehicle behavior, vehicle nose-down tilt attitude, control stick position, and power requirements as functions of airspeed. Airspeed was measured by pacing the vehicle with an automobile with calibrated speedometer and limiting tests to conditions of negligible winds and/or correcting for wind in making airspeed calculations. Tests for record were made in winds of 3 mph or less. This pace car technique was adopted after attempts to use on-board airspeed indicating equipment failed to give satisfactory levels of accuracy.

Vehicle attitude and control stick position data was obtained through the telemetry system.

Engine power data was accumulated by photo techniques recording fuel flow, exhaust gas temperature and turbine speed. A second index of power consumed was the collective pitch data continuously recorded through the telemetry system. Lower speed runs (up to approximately 25 mph) were made without the vertical fin. Higher speed runs were made with the fin in place.

Control stick position data as a function of airspeed is presented in Figure 45. Superimposed on this plot, for comparison purposes, are the predicted values. Excellent agreement with the estimated data will be noted.

Pitch attitude data as a function of airspeed is presented in Figure 46. This data contains considerable scatter, due mainly to the restricted runway length available (.3 mile). This short space made it difficult for the pilot to accelerate to the test speed, stabilize all conditions for recording, and then decelerate. It appears that the predicted values for pitch attitude as a function of forward speed are reasonable. Photo coverage substantiates this, as shown in Figure 47.

Power required data as a function of airspeed is presented in Figure 48. It can be seen that the power required is reduced with increasing speed (to the limit of the testing accomplished). However this tendency is quite flat compared to the shart drop in power required for conventional rotorcraft. This difference can be attributed primarily to the high disc loading and its attendant high inflow velocities.

A maximum airspeed of 44 mph was achieved during these tests. It is stressed that the maximum speed of 44 mph attained is not the maximum speed capability of the vehicle but rather the maximum speed which could be attained under the test conditions. Runway length limitations restricted maximum speed runs to those presented here. It is the opinion of the pilot that speeds considerably in excess of those attained can be achieved with a less restricted environment.

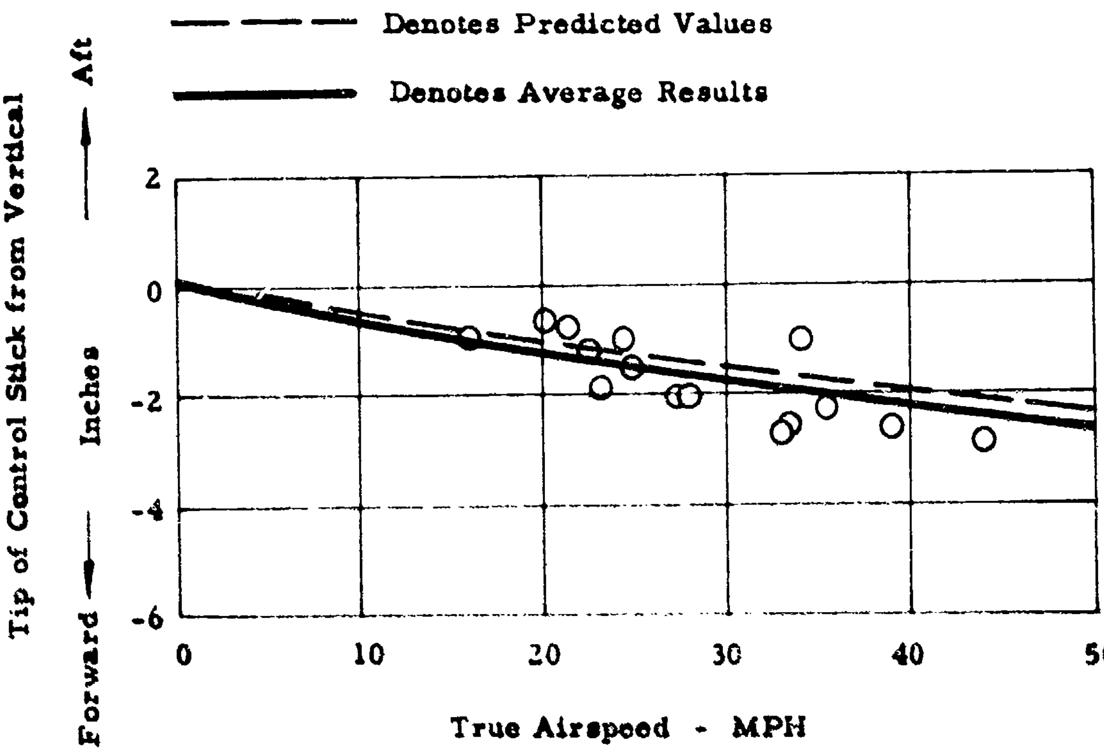


Figure 45 PITCH STICK POSITION vs AIRSPEED

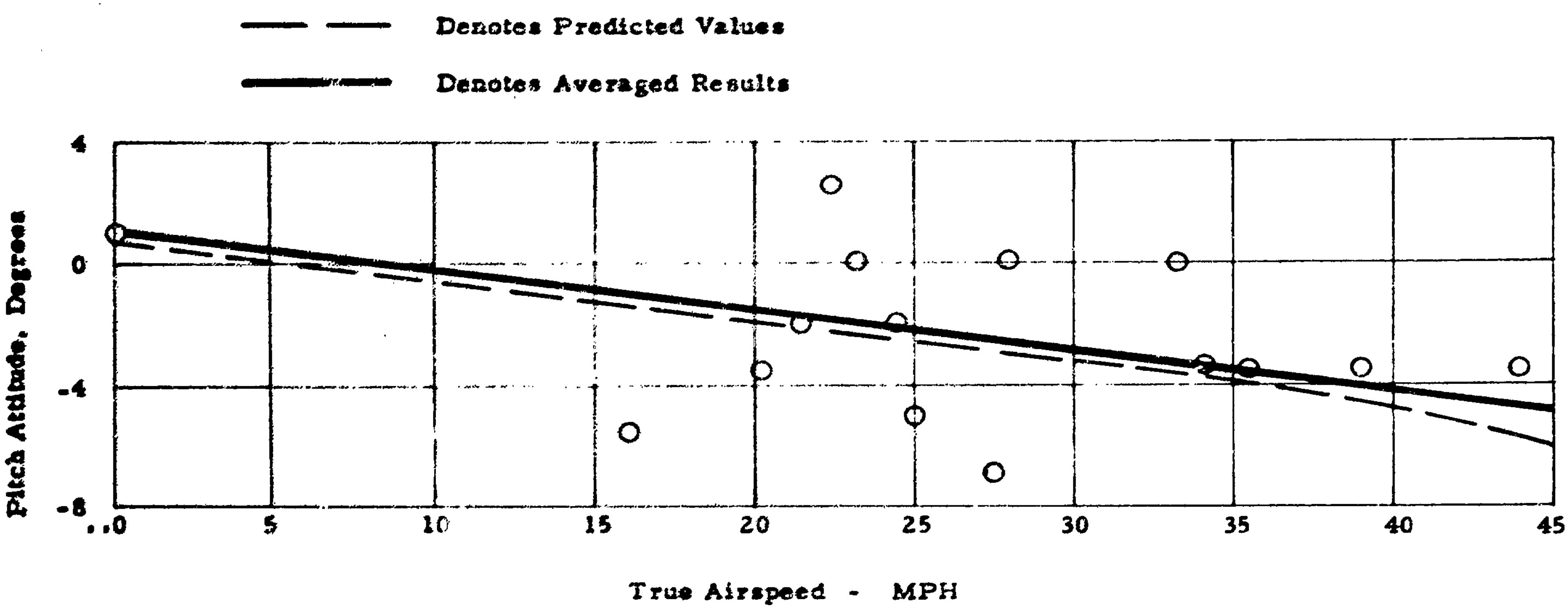


Figure 46 VEHICLE PITCH ATTITUDE vs AIRSPEED

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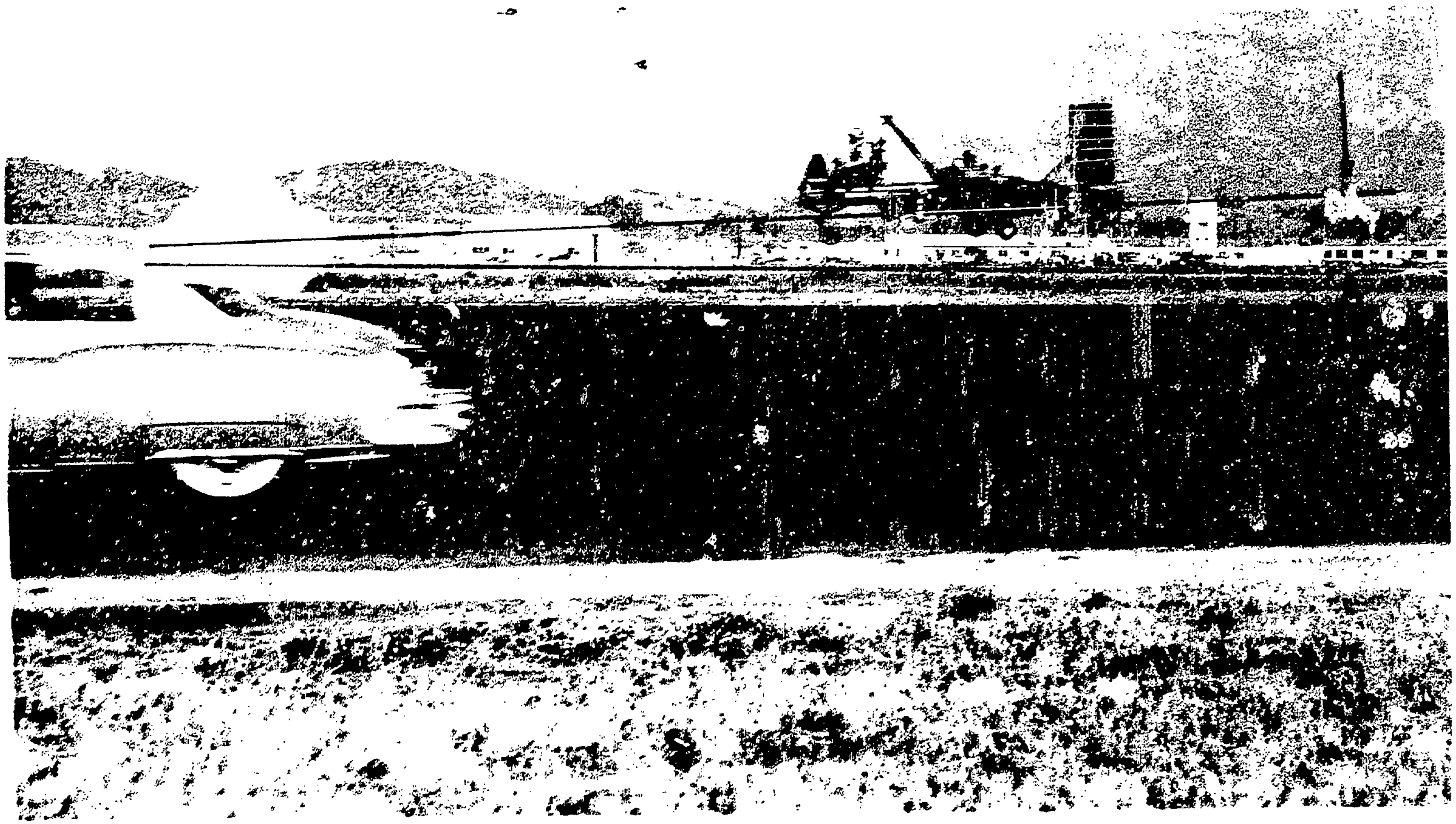


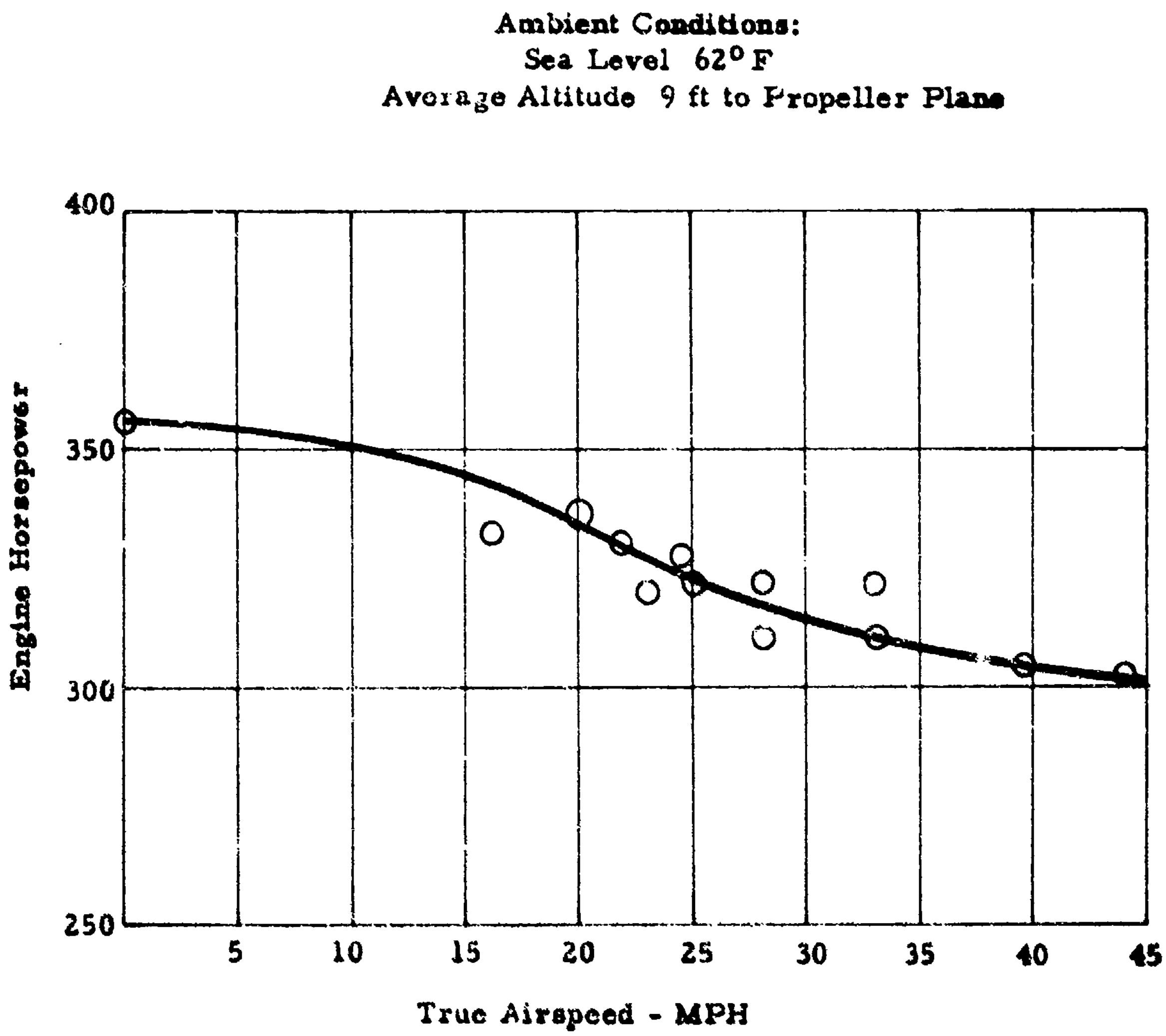


Figure 48 POWER REQUIRED vs AIRSPEED

Ground Effect Tests

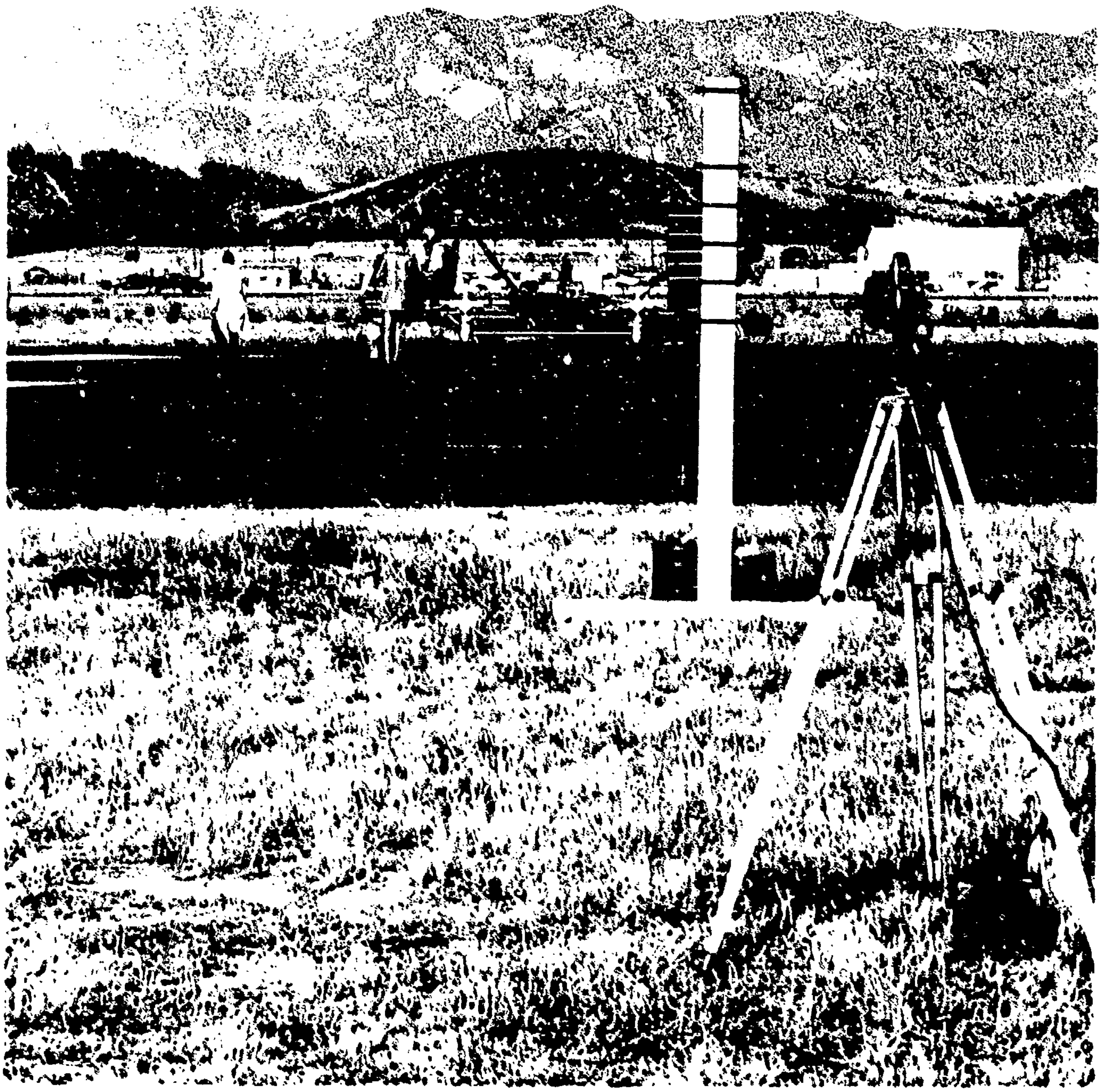
To determine the ground effect on power required for hovering, a test was conducted utilizing the set-up illustrated in Figure 49. The vehicle was flown at four predetermined altitudes. Altitude was carefully "measured" by external photo means using a graduated altitude "scale" with pre-calculated camera, scale and vehicle positions. Engine power data was recorded by on-board photo means as previously described under speed tests. A typical altitude flight is shown in Figure 50.

Power required to hover at various altitudes is presented in Figure 51. From these same tests, the variations in propeller thrust vs blade angle as a function of altitude have been plotted in Figure 52. A further plot of ground effect is presented in Figure 53, which shows thrust variation with altitude.

Weight Lifting Tests

The vehicle's weight lifting capabilities and the effect of gross weight on horsepower required to hover were investigated by a series of weight lifting runs wherein the gross weight was varied by installation of lead pigs on the cargo deck and by variations in fuel load. These runs were designed to be the free flight counterpart of tied-down thrust measurements. As such, the pilot was instructed to attain and hold an altitude of 6 to 12 inches with each gross load. Successively larger ballast loads and up to 2900 lbs gross weight (500 lbs over normal gross weight) were lifted.

The results of these tests are presented in Figures 54 and 55. Figure 54 shows the relationships between power and gross weight to lift the vehicle just clear of the ground (6 to 12 inches altitude). Figure 55 presents the relationships between propeller blade angle and propeller thrust for the same 6 to 12 inches altitude conditions.



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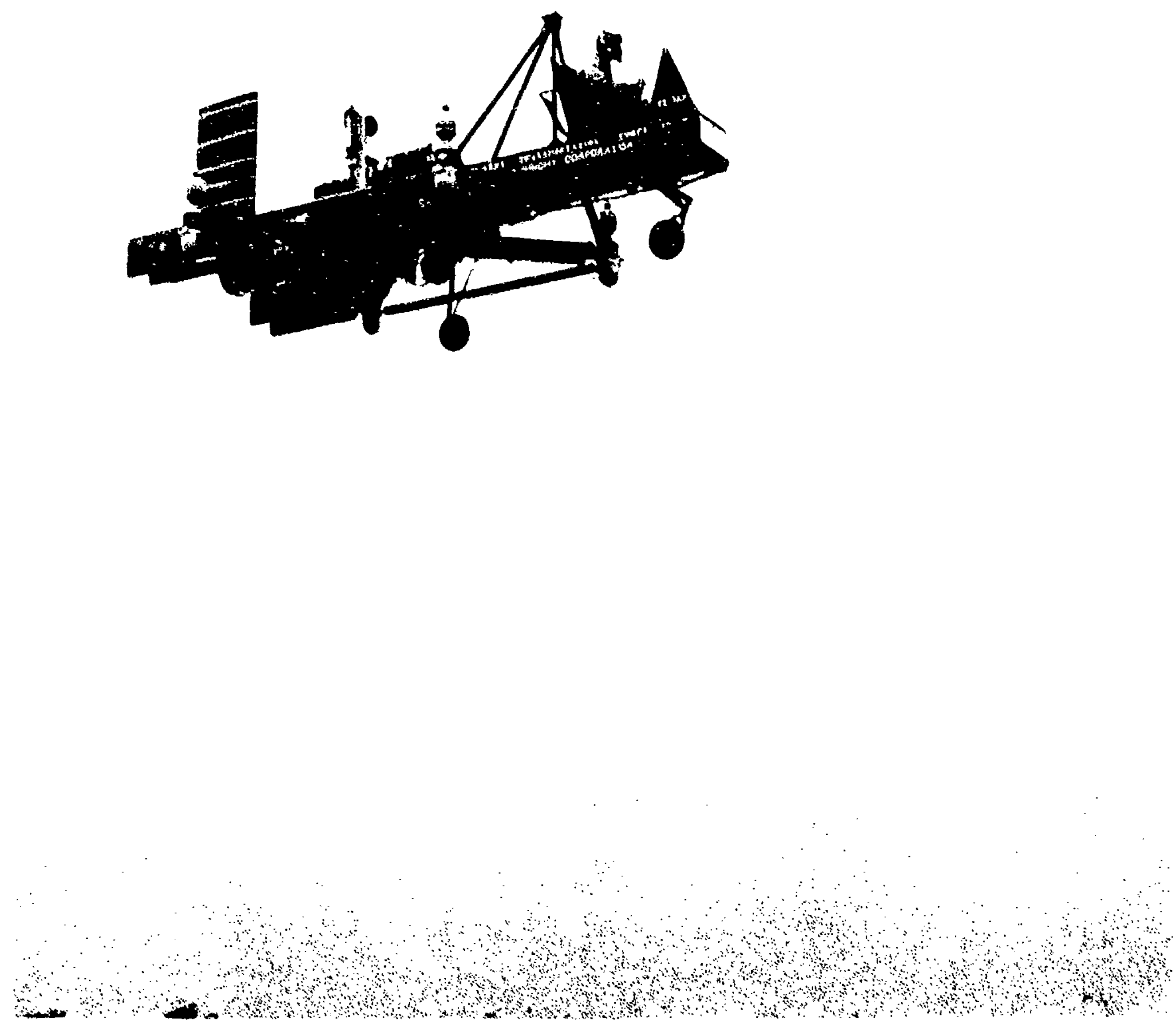

Figure [illegible] FREE FLIGHT AT APPROXIMATELY 25 ft ALTITUDE

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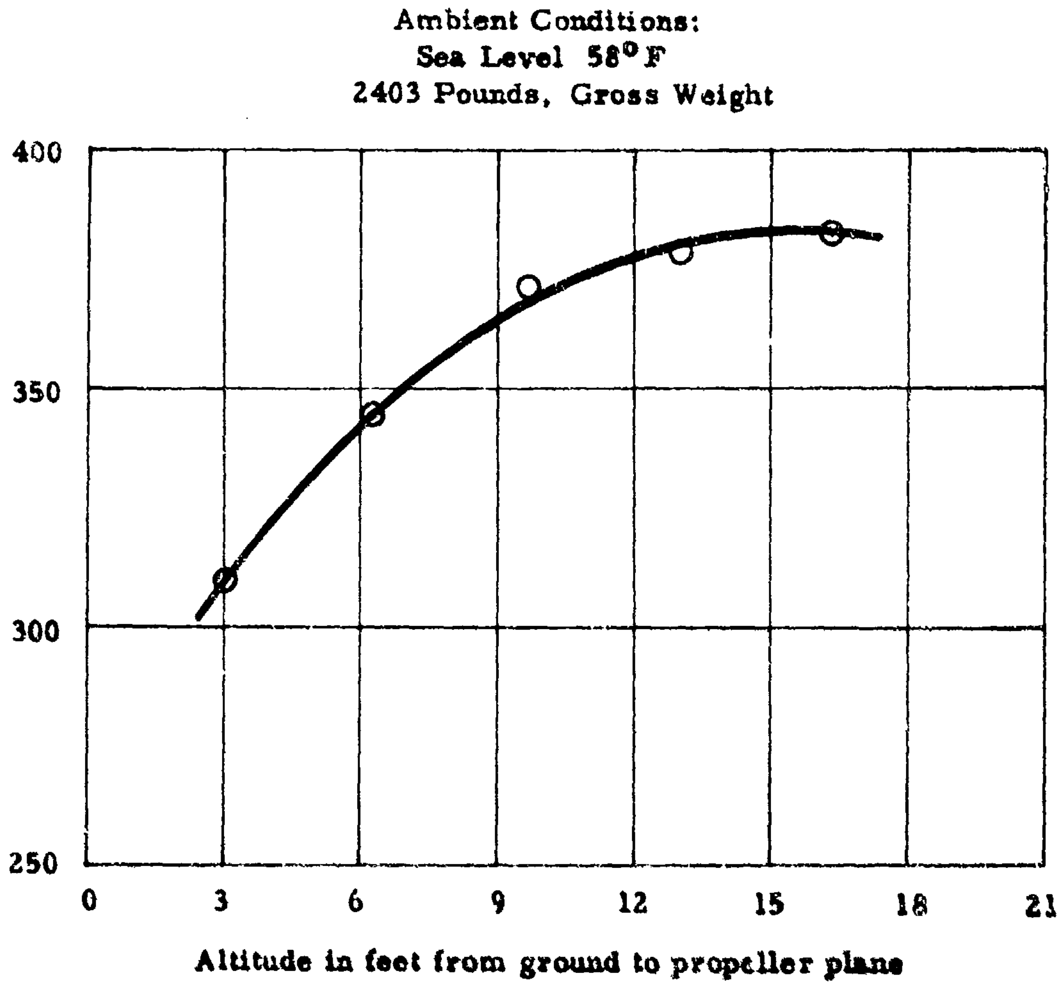


Figure 51 POWER REQUIRED vs ALTITUDE, HOVERING

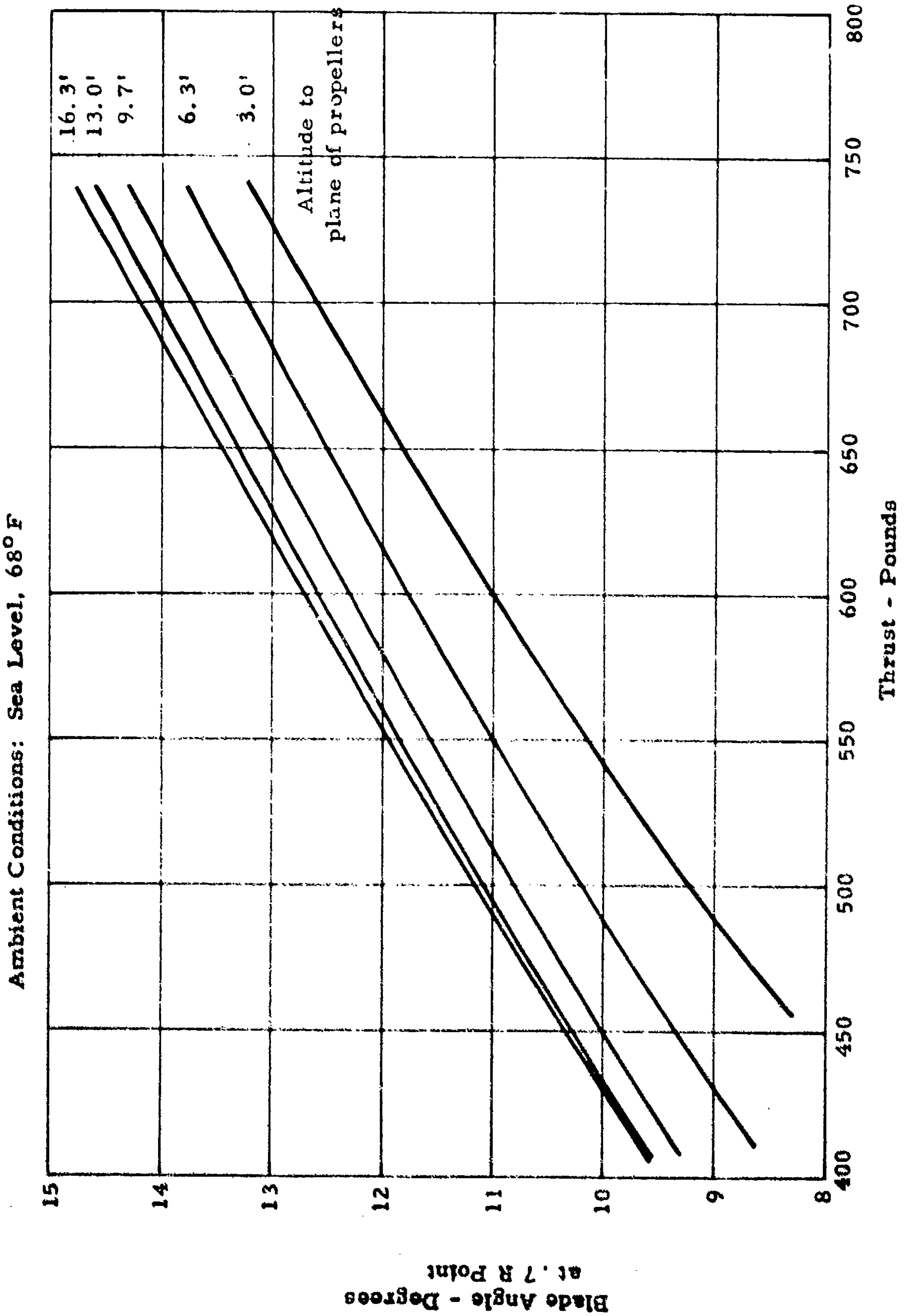


Figure 52 PROPELLER THRUST vs BLADE ANGLE, VARYING ALTITUDES

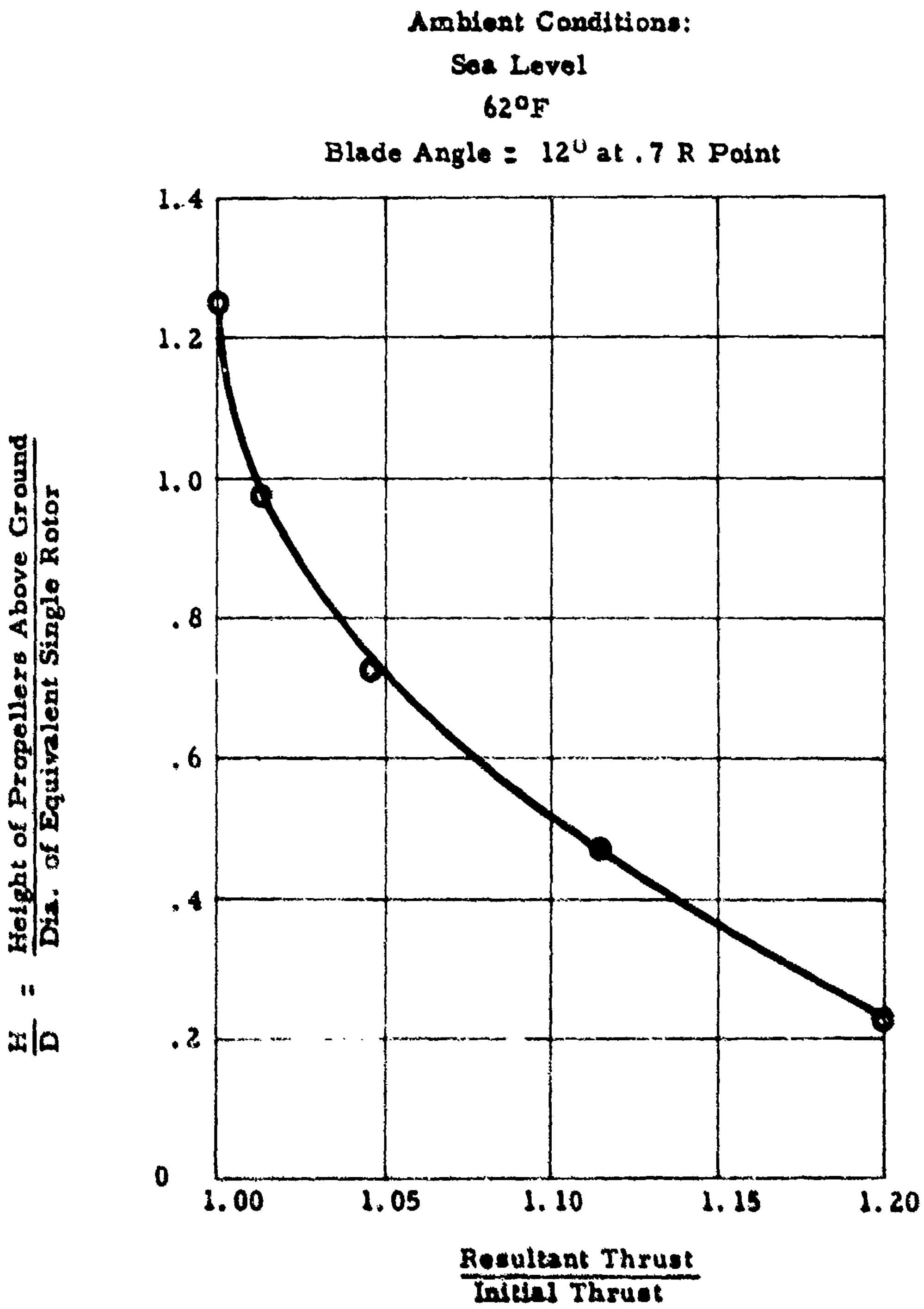


Figure 53 PROPELLER THRUST vs ALTITUDE FOR CONSTANT BLADE ANGLE

Ambient Conditions:
Sea Level
58°F

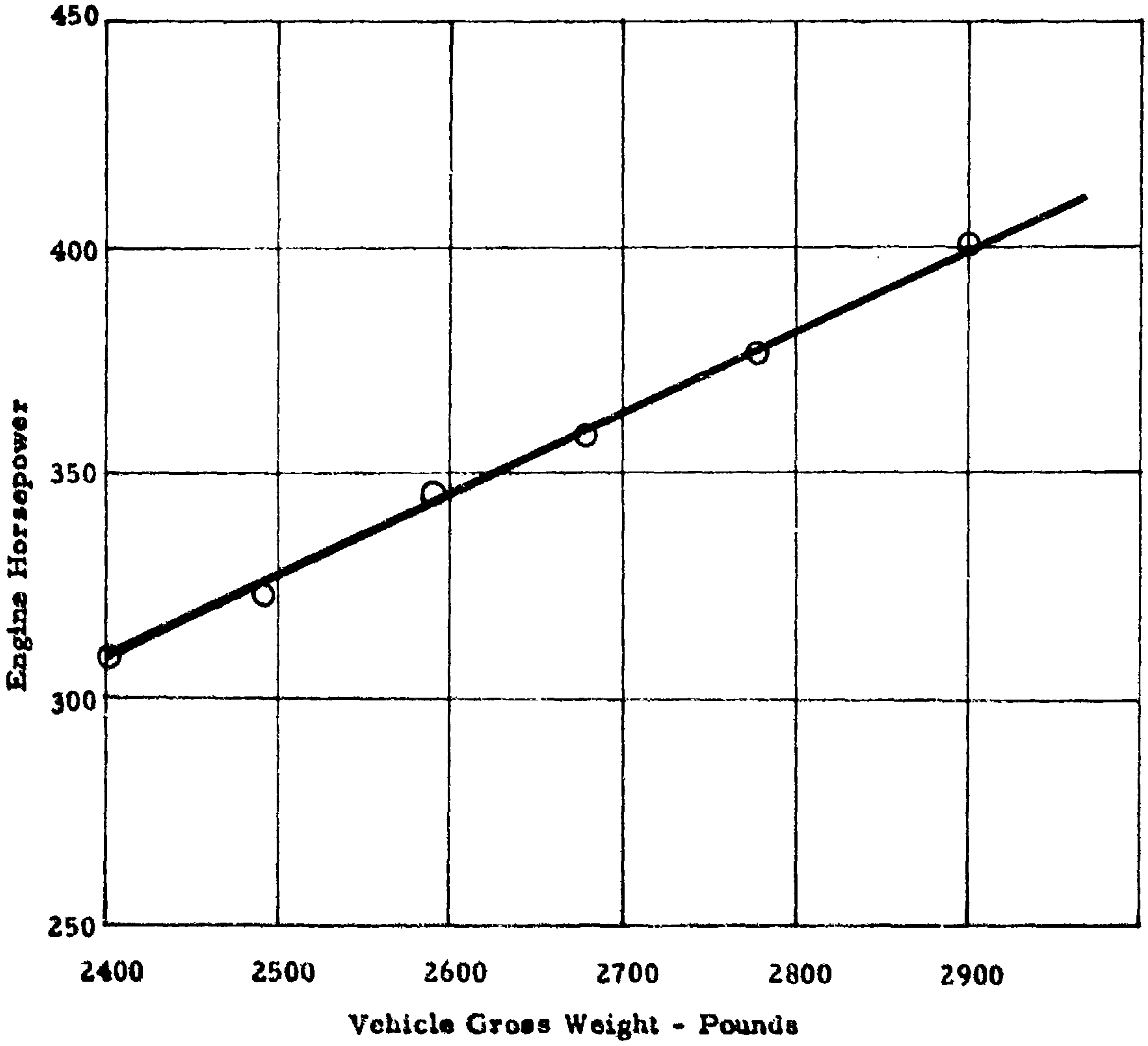


Figure 54 POWER REQUIRED AT LIFTOFF vs GROSS WEIGHT

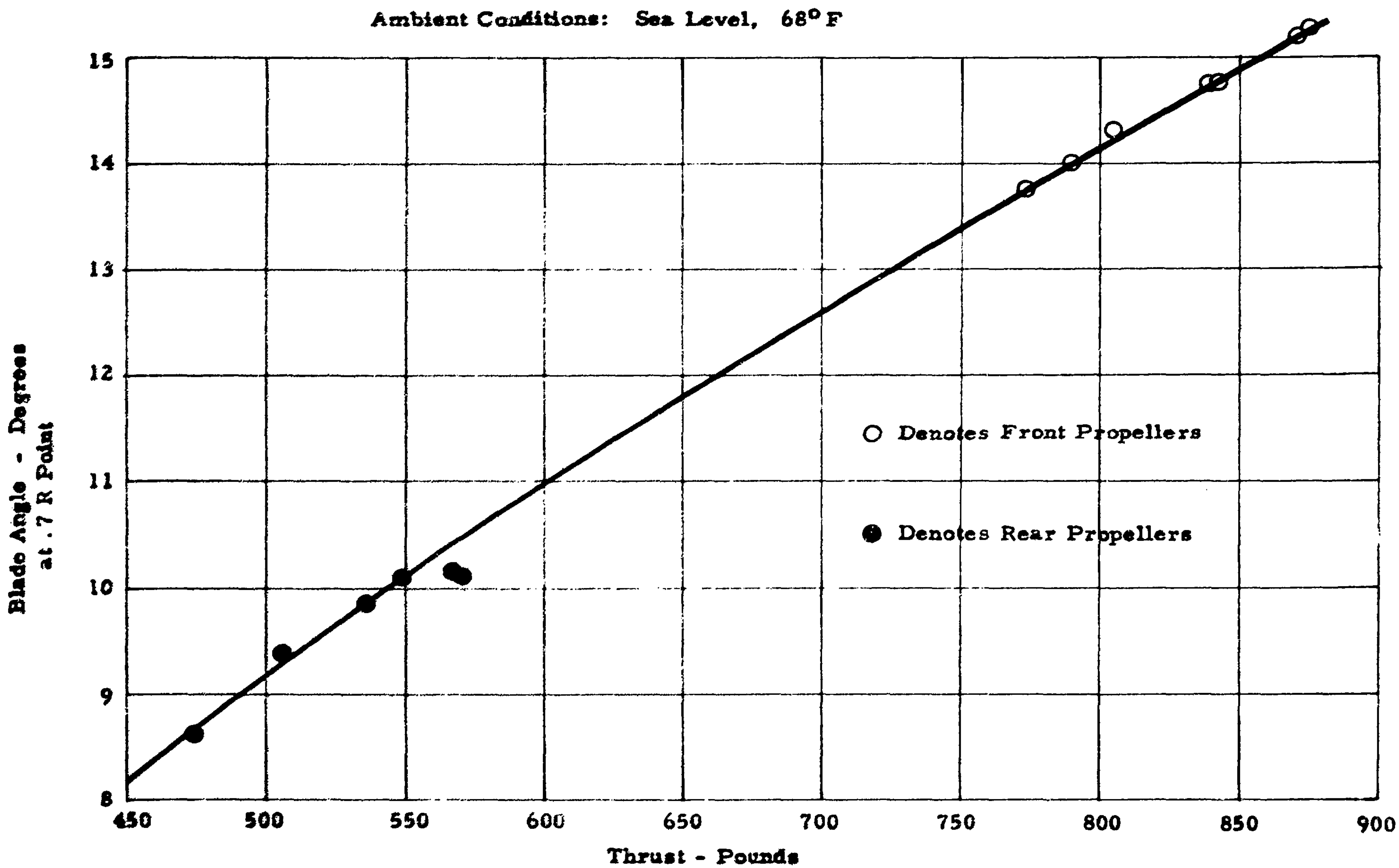


Figure 55 PROPELLER THRUST vs BLADE ANGLE AT LIFTOFF

Miscellaneous Testing

During the course of the principal free flight testing described above, other tests of secondary importance were accomplished and are considered to be significant:

a. Flights over Unpaved Surfaces
b. Flights Close to a Building
c. Flights with a second crewman
d. Military Demonstrations
e. Flights with Guard Rings

Flight Tests Over Unpaved Surfaces

In order to determine the dust raising characteristics of the Aerial Platform with its high disc loading (18 lbs per square foot), a brief series of flights were made off of the paved runways normally used.

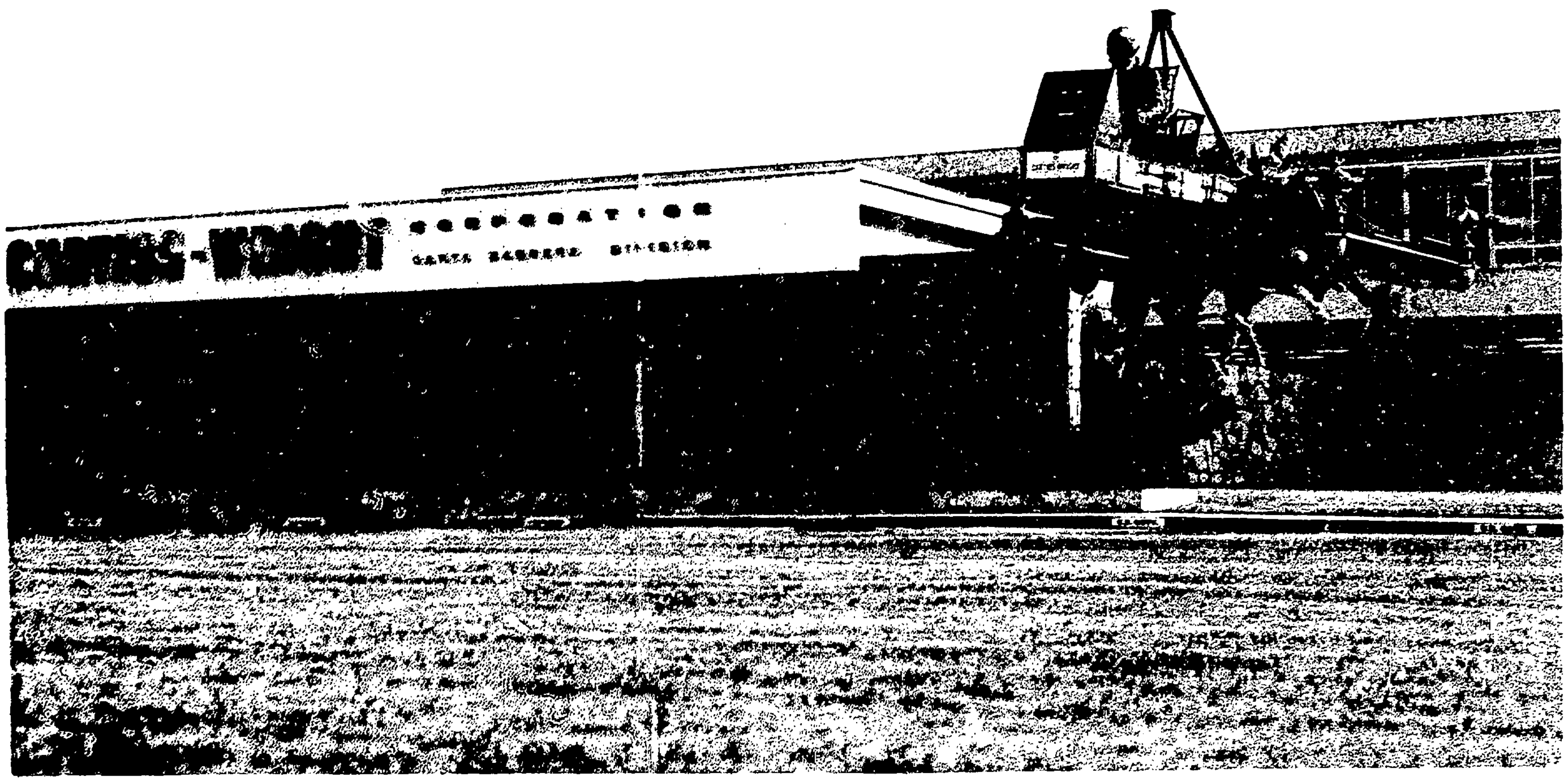
A single test flight at approximately 4 foot altitude over unpaved areas adjacent to the airport runways produced a sizeable dust cloud and reduced pilot vision very noticeably. The dirt in this particular location was very soft and dry, like talc, making this test very severe. This test was repeated in the same area after rainfall had packed the soil. No dust or debris was raised.

A single test flight at approximately 4 foot altitude over a grassy plot (Bermuda grass) produced almost no dust and was considered to be quite satisfactory. It is believed that landing and take-off from the grassy surface would present no hazards.

Free Flight Tests Close to a Building

Free flight testing in close proximity to a building was accomplished to establish if the flight characteristics of the vehicle would be affected by backwash from the building. See Figure 56 The pilot flew approximately along the middle of a 35 foot roadway giving a propeller-to-building clearance of approximately 20-25 feet. He reported "some backwash" from the building but not enough to concern him. Altitude on 6 "passes" varied from 2 feet to 8 feet.

9413



F[illegible] [illegible] [illegible] CLOSE [illegible] [illegible]ILDING

Free Flight With Second Crewman

On three occasions, the VZ-7AP was flown with a second crewman on board. These runs permitted flight test and engineering personnel to observe first hand the behavior of the vehicle.

The first such two man flight was made with the second crewman seated in the normally unused co-pilot seat. On this flight the full complement of instrumentation was on board, giving a then record gross weight of 2545 lbs. This test configuration produced the furthest forward center of gravity location in our entire test program. The pilot experienced no difficulty in flying with this load change.

Subsequent two man flights were made, using a special jump-seat installed on the main cargo deck in lieu of the instrumentation package. Gross weight for these flights was 2446 lbs. The frontispiece of this report depicts one of these later two man flights.

Military Demonstrations

On two separate occasions, the Aerial Platform was demonstrated in flight for military observers.

The first demonstration, October 28, 1959, was accomplished at the Santa Barbara Municipal Airport, and was attended by representatives from all branches of the military. In this demonstration, the pilot performed a broad variety of hovering and translational maneuvers, first with an open load deck and later with a 57 mm recoilless rifle mounted on the cargo deck. The pilot experienced no detectable changes in control action as a result of flying with this sample military load. The demonstration flight is shown in Figure 57.

The second military demonstration was conducted at Ft. Ord, California on December 18, 1959, for the staff of the Combat Development Experimentation Center. This demonstration was conducted with the normal complement of telemetering and instrumentation equipment on board.

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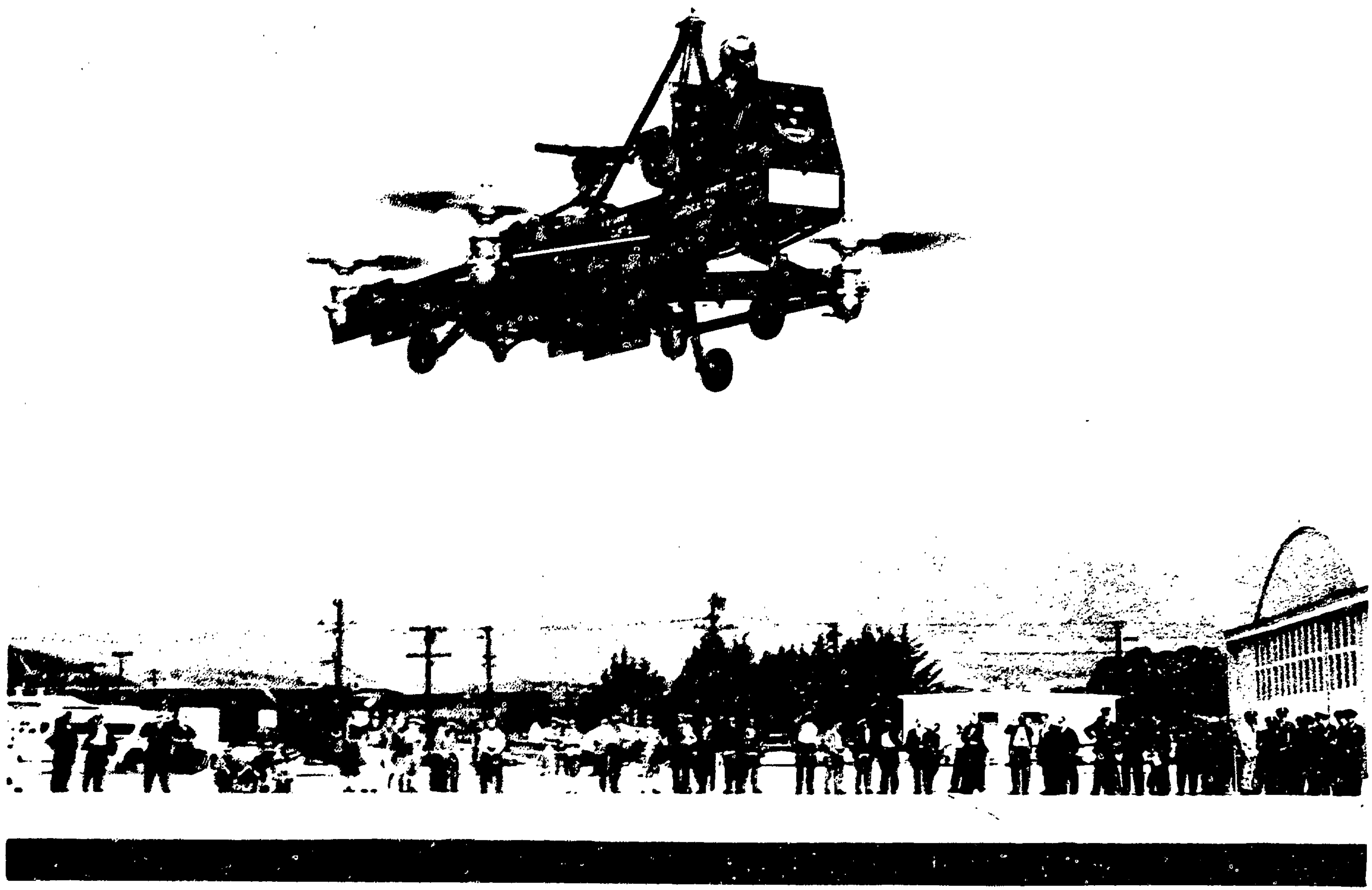


Flights with Guard Rings

The VZ-7AP vehicle was flown in most cases without the use of the guard rings surrounding the propellers. The rings are shown in Figure 16 . This deletion was primarily for convenience in towing the vehicle on the road, entrance and exit from buildings and for easier ground handling in general.

The effect that the guard rings might have on the flight characteristics of the vehicle were assessed with a series of test flights with guard rings in place. The tests included hovering flight and translation in all directions. Based on pilot comments, motion picture film, and eyewitness reports, the rings did not alter the flight characteristics for speeds up to approximately 15 mph. The pilot's comments on tests at approximately 20 mph indicated a slight tendency for the vehicle to porpoise. Review of the wind tunnel data suggests that such possibility exists, for the guard rings are shrouds of sorts, and do alter the pitching moment in translation over that for the bare propellers alone.

MODIFICATIONS

Modifications

The flight testing of the Aerial Platform for data purposes, as described above, was conducted in essentially one basic configuration. There were, however, several important modifications made to the vehicle in arriving at the test configuration. These modifications can be grouped as follows:

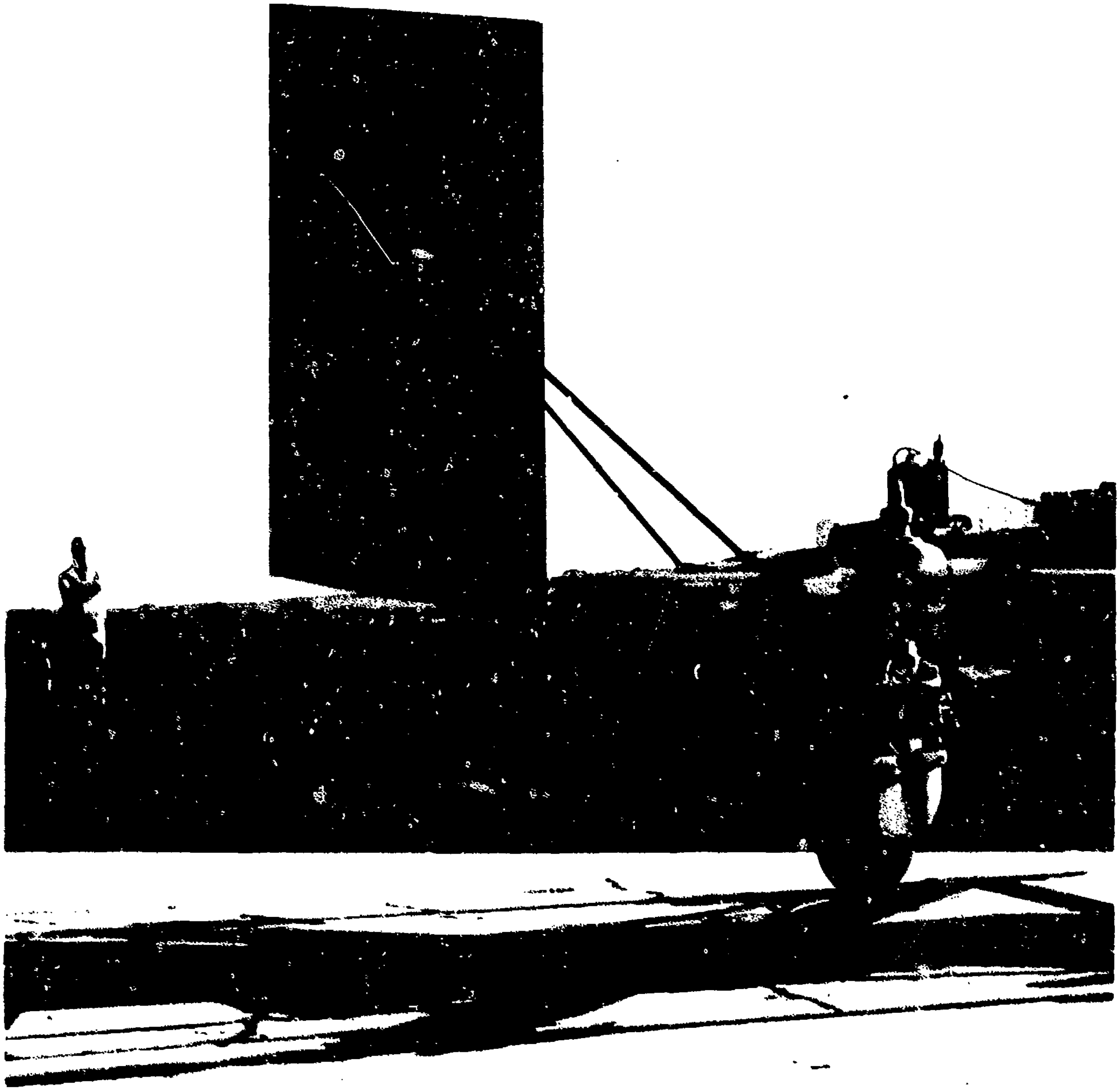
a. Yaw Stability and Control Modifications
b. Control Sensitivity Modifications
c. Weight Reduction Modifications
d. Propeller Rotation Modifications

Yaw Stability and Control Modifications

As a result of the limited yaw control effectiveness experienced in our first free flight tests, a program to improve the yaw control was undertaken.

Yaw vanes were added under the rear inboard portion of the two rear propellers. These rudders are pivoted on fore and aft horizontal axes and driven by push-pull tubes from the rudder cable quadrant mounted over the engine tail pipe. See Figure 34 . These yaw vanes were designed to provide major improvements in yaw control moments available and secondarily to provide a directional stabilizing tendency to help offset the long nose of the test vehicle. The effectiveness of these yaw vanes can be seen in Figure 36.

A relatively large fixed fin was developed for use in the higher speed tests. This fin was designed to provide ample directional stability in the nose down-high speed yaw-roll coupling mode. The fin is shown in the 3 view drawing, Figure 15, and by photographs in Figures 47 and 58 . The net stabilizing effect with this fin in place can be seen in Figure 38.



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Control Sensitivity Modifications

Although the control system had been designed to provide nearly equal control stick sensitivity in both pitch and roll directions, early flights indicated that the control was overly sensitive in the roll direction. Telemetered records indicated that the pilot's normal roll commands amounted to only ± 1.0 inches of lateral stick motion. This small motion made it quite easy to over control the vehicle in roll.

A change was made in the mechanical mixer in the control system to alter the geometry of the linkages and reduce roll sensitivity by approximately 50% with no apparent change in pitch sensitivity.

Weight Reduction Modifications

In order to provide a broader margin of thrust over weight for the more taxing tests such as weight lifting, speed runs, etc., a program of lightening the vehicle was accomplished. A total of 125 lbs was removed by reduction of size of battery (22 lbs savings), removal of all stability augmenting equipment (60 lbs savings), replacement of overturn structure with one of thinner material (26 lbs savings) and a number of lesser items.

This reduction of dead weight items permitted flight testing with a considerably broader variation in vehicle gross weights. Minimum take-off weight was 2263 lbs. Maximum take-off weight was 2900 lbs. See Table 6 for a tabulation of weights and center of gravity locations for all flight tests.

Propeller Rotation Modifications

During the course of the early flights a tendency to develop a roll oscillation during forward flights of approximately 20 mph was observed. This was traced to the particular directions of rotation of the propellers, wherein yaw reaction from a roll maneuver was opposite in direction to the roll maneuver, i.e., left roll was producing a right yaw reaction and vice versa. See Figure 59.

The characteristic interchangeability of components designed into the gear system permitted the outright interchange of the right hand propeller gear boxes with the left hand boxes, to accomplish an effective reversing of propeller rotations. Study of the gearing schematic, Figure 20, will show that exchange of the so-called "gear up" case and "gear down" case (i.e., driven gear location in relation to the driven pinion) will reverse the rotation of the propeller shafts. The propellers were exchanged integrally with their respective gear boxes.

Subsequent testing up to 44 mph showed that the involuntary roll oscillation tendency in forward flight was eliminated by this change.

Final Configuration

The final configuration of the VZ-7AP Aerial Platform at the conclusion of the 32 hour flight test program is presented in Figure 60.

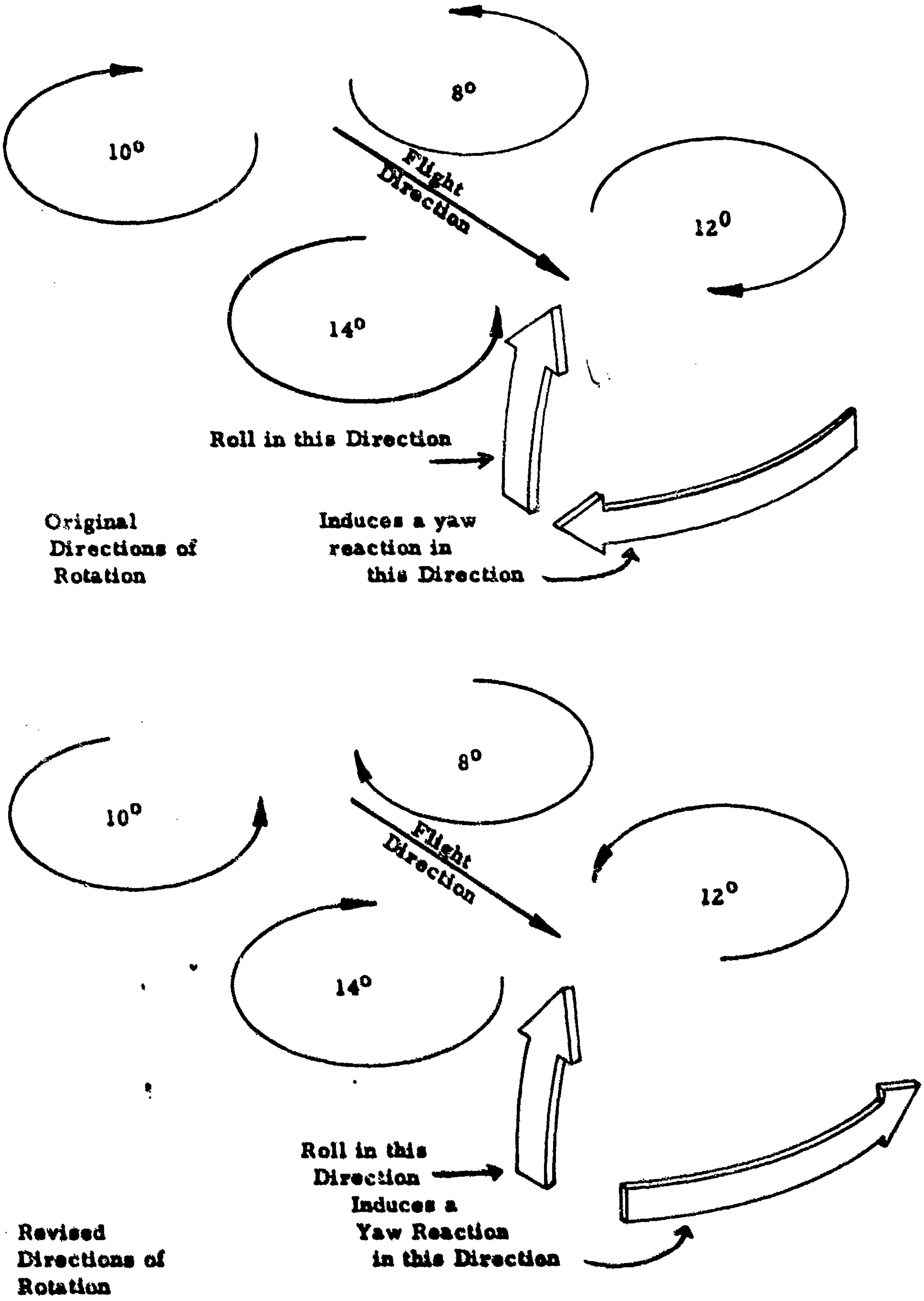

Figure 59 PROPELLER ROTATION MODIFICATIONS

TABLE 6

FLIGHT TEST WEIGHT & BALANCE SUMMARY DATA

No.	CONFIGURATION DESCRIPTION	GROSS WT	C. G. LOCATION LONG.	VERT.
1	Initial Free Flight Configuration with Instrumentation	2410.	187.9	34.2
2	(1) with 4 Yaw Vanes	2456.	188.5	34.0
3	(2) after Weight Reduction, without Instrumentation	2268.	188.4	32.4
4	(3) with 57 MM Gun	2367.	188.7	33.5
5	(4) with Guard Rings, without Gun	2298.	188.5	32.4
6	(3) with Instrumentation without Gun or Rings	2335.	188.7	32.7
7	(6) with Fin	2348.	189.1	32.9
8	(6) with 2nd Crewman, Increased Instrumentation without Fin	2545.	186.9	34.2
9	(6) with Maximum Instrumentation & Fin	2403.	187.9	33.1
10	(9) with Lead Ballast	2900.	190.6	34.3
11	(3) with 2nd Crewman without Voice Radio	2446.	188.3	34.2

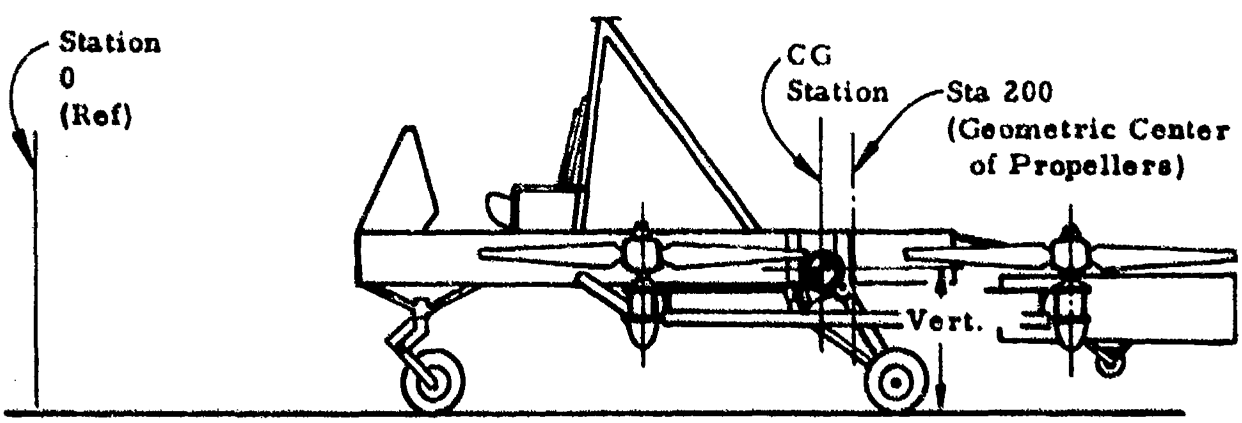

TABLE 7

TEST SUMMARY

Performance Summary

Total Number of Flights	102
Maximum Speed	44 mph
Maximum Altitude	25 ft
Typical Yaw Rate	65^{o}/sec
Typical Yaw Acceleration	$12^{o}/sec^{2}$/inch pedal displ.
Typical Yaw Deceleration	$23^{o}/sec^{2}$/inch pedal displ.
Typical Roll Rate	52^{o}/sec
Typical Roll Acceleration	$20^{o}/sec^{2}$/inch stick displ.
Typical Pitch Rate	46^{o}/sec
Typical Pitch Acceleration	$22^{o}/sec^{2}$/inch stick displ.
Design Gross Weight	2400 lbs
Design Useful Load (incl. payload)	647 lbs
Maximum Gross Weight Flown	2900 lbs
Maximum Useful Load (incl. payload)	1147 lbs

Summary of Test Times

Ground Tests	9 hrs. 59 mins.
Tether Tests	2 hrs. 52 mins.
Flight Tests	19 hrs. 9 mins.
Total Engine Time	32 hrs. 0 mins.
Actual Time in Air	11 hrs. 54 mins.
Longest Single Flight	11 mins. 22 sec.

MAINTENANCE CHARACTERISTICS

The VZ-7AP Aerial Platform is a simple and rugged device which has proven in testing to be free of many of the relative fragilities of conventional rotary wing aircraft.

The Aerial Platform has demonstrated very favorable characteristics in regards mechanical reliability. At the conclusion of the aforementioned flight test program the vehicle was still flying on the original engine, the original propellers, the original drive shafts, the original gear boxes, bearings and most of the original gears. (Three propeller gear box pinions were exchanged due to lubrication and engine surge problems prior to optimization of the lubrication and engine control systems.) These successes are due to the soundness of the basic concept and to the attention paid to details in the design development of this research test bed vehicle.

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